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EXTENDED APOLLO SYSTEMS UTILIZATION STUDY

Final Report
Volume 2
Mission and Performance Analysis
(U)

16 November 1964

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FOREWORD

This volume constitutes a portion of the Apollo X Study final report (SID 64-1860) prepared as part of the Extended Apollo Systems Utilization Study conducted by the Space and Information Systems Division of North American Aviation, Inc., for the National Aeronautics and Space Administration's Manned Spacecraft Center under Contract NAS9-3140, dated 6 July 1964. S&ID acknowledges the outstanding technical contributions made to the study by a number of companies; these organizations are identified below along with the title of the report for which they are responsible.

The final report has been prepared in the series of 23 volumes listed below:

1. Summary
2. Mission and Performance Analysis
3. Experimental Programs
4. Configurations, Structures, and Weights
5. Mission Plans and Functions
6. Environmental Control System (AiResearch)
7. Fuel Cells (Pratt & Whitney)
8. Alternate Power Source (General Electric)
9. Cryogenic Storage System (Beech Aircraft)
10. Service Module SPS Engine (Aerojet)
11. Command Module RCS Engine (Rocketdyne)
12. Service Module RCS Engine (Marquardt)
13. RCS Propellant Tanks (Bell)
14. Guidance and Navigation (A-C Spark Plug)

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15. Alternate Guidance System (Autonetics)

16. Guidance Computer (Raytheon)

17. Stabilization and Control System (Honeywell)

18. Communications and Data System (Collins Radio)

19. Earth Landing System (Northrop-Ventura)

20. Subsystems Supplement

21. Reliability and GSE

22. Development Planning

23. Condensed Summary

TECHNICAL REPORT INDEX/ABSTRACT

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APOLLO X

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*APOLLO X, *LUNAR MAPPING MISSION, *LUNAR AND EARTH POLAR ORBITS, *BEHAVIORAL EXPERIMENTS, *BIOMEDICAL EXPERIMENTS, *ENGINEERING EXPERIMENTS, *PHYSICAL SCIENCE EXPERIMENTS, *APOLLO SUBSYSTEMS LIFE EXTENSION.

ABSTRACT

This 23-volume report defines in depth the design and operational characteristics of a vehicle based upon the identical subsystems approach employed in the Concept 0 configuration, which was defined in the Extended Mission Apollo Study (contract NAS9-1963). The maximum mission duration is arbitrarily limited to 45 days. This vehicle, identified as Apollo X, appears to represent the most logical next-step in manned spacecraft capability extension beyond the current Apollo and Gemini programs.

The primary objective of the Apollo X study has been to define a "standard" spacecraft design capable of alternately performing extended-duration lunar or earth orbital missions of NASA near-term interest. Included in this primary objective were studies to determine specific modifications required for each subsystem to accomplish the extended missions. Subsystem qualification test programs, which would substantiate the analytically derived extended-life subsystems, were also defined. In the experimental area, emphasis was placed upon the definition and integration of experimental packages (for earth orbital missions) based upon experimental lists provided by NASA at the outset of the study. All development-factors investigations were based on an integrated, but non-interference, relationship with the current Apollo program.

The Apollo X study was initially focused toward consideration of earth orbital experimental missions of up to 35 days' duration. Secondary studies were aimed at determining the required modifications of the 45-day earth orbital CSM design when applied to a lunar survey mission (including up to 28 days in lunar polar orbit). Several weeks after study initiation, however, NASA directed that equal emphasis to be given to the lunar mapping mission and requested that primary emphasis be placed on the definition of a standard CSM suitable for both the lunar and earth orbital missions. Accordingly, a reevaluation was made of previously developed earth orbital mission requirements. This reevaluation indicated that the lunar survey mission imposed overriding requirements in most subsystem areas.

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SID 64-1860-2	2.	Mission and Performance Analysis	S&ID
SID 64-1860-3	3.	Experimental Programs	S&ID
SID 64-1860-4	4.	Configurations, Structures, & Weights	S&ID
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SID 64-1860-6	6.	Environmental Control System	AiResearch
SID 64-1860-7	7.	Fuel Cells	Pratt & Whitney
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SID 64-1860-9	9.	Cryogenic Storage System	Beech Aircraft
SID 64-1860-10	10.	Service Module SPS Engine	Aerojet
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SID 64-1860-19	19.	Earth Landing System	Northrop-Ventura
SID 64-1860-20	20.	Subsystems Supplement	S&ID
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SID 64-1860-22	22.	Development Planning	S&ID
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PREFACE

The NASA Manned Spacecraft Center has for the past several years examined the application of the Apollo spacecraft to missions alternate to the basic lunar landing mission for which it is currently designed. The overall objective of these studies is an assessment of the advantages and disadvantages associated with the application of developed hardware to these other potential missions.

NASA studies of near-term applications, which represent possible initial extensions to current Apollo capabilities, include use of the Apollo spacecraft as (1) a space station logistics resupply vehicle (carrying up to six men), (2) an earth-orbital space station or experimental laboratory, (3) a lunar mapping and survey vehicle, etc. This report presents the results of the most recently concluded studies of the Extended-Mission Apollo, in which the application to a 45-day earth orbital laboratory role and to an extended lunar orbit mission have been examined in depth.

The initial Extended-Mission Apollo study, initiated in August 1963 under contract to NASA/MSC, examined the suitability of Apollo as an earth-orbital biomedical/behavioral experimental laboratory. These experiments were to provide a basis from which man's suitability for protracted space missions could be determined. Three basic configuration concepts, indicated in Figure 1, were investigated throughout the study. Configuration Concept I utilizes only the Apollo command and service modules (CSM), with experimental in-orbit work space made available in the command module by the elimination of one crewman from the current crew size of three. The Apollo CSM subsystems were modified to sustain orbital operations for periods of up to 120 days without resupply. Configuration Concept II consisted of the Apollo CSM plus a 5600 cubic foot laboratory module built within the geometric limits of the LEM adapter. In this concept, the CSM subsystems support the laboratory module functions for the 120-day resupply period. Consequently, Concept II has a laboratory which is dependent on the CSM subsystems. The third configuration (Concept III) is similar to Concept II with respect to the addition of the separate laboratory module within the LEM adapter section; however, the Concept III laboratory was designed to carry its own subsystems; therefore, it is considered to be independent of the CSM subsystems. The subsystems in the CSM-independent laboratory module were designed for one-year continuous operation with resupply of expendables by the Apollo CSM as required. Included as a part of the original Extended-Mission Apollo study was the establishment of detailed development plans

Figure 1. System Evolution

including costs, schedules and manufacturing, and facilities plans. These development plans were based on an integrated but noninterference relationship with the Apollo program and on maximum utilization of Apollo technology, facilities, etc. The laboratory module, for example, utilized a large percentage of Apollo/Saturn interstage tooling.

Results of the initial Extended-Mission Apollo studies revealed several additional factors that warranted further investigation. The required 120-day resupply cycle dictated the use of advanced subsystem concepts in several areas. For example, the existing fuel cell electrical power system in the Apollo was replaced with a solar cell-battery power system to eliminate the excessive weight and volume penalties attributable to the fuel cells and associated expendables. Additionally, a molecular sieve was employed in the environmental control system to eliminate the large weight and volume attendant with using the existing Apollo lithium hydroxide CO_2 removal system for 120 days. Consequently, the limiting mission duration which might result from using only current Apollo subsystem concepts was not known. Also, although the feasibility of extending the life of the current Apollo subsystems had been established in the initial study, specific techniques for accomplishing this life extension had yet to be determined.

For these reasons, further studies were initiated as an addendum to the Extended-Mission Apollo contract. The purpose of these additional studies was to define the design characteristics and the maximum earth orbital mission duration of the Apollo CSM, assuming restriction to use of existing Apollo subsystem concepts. This configuration was identified as Concept O. Included in this study was examination of each subsystem to determine the life extensions possible through the addition of spares and redundancies. The study concluded that the earth-orbital duration capability of Concept O was approximately 90 days (based on Saturn IB payload limits).

After review of these findings, NASA initiated the current Extended Apollo Systems Utilization Study, one part of which is to define in depth the design and operational characteristics of a vehicle based on the identical subsystems approach employed in Concept O, but limited arbitrarily to a 45-day maximum mission duration. This vehicle, identified as Apollo X, appears to represent the most logical next-step in manned spacecraft duration capability beyond the current Apollo and Gemini programs.

As part of this same study program, the characteristics of the CSM configurations associated with the dependent and independent laboratory module (as in Concepts II and III) are being investigated relative to earth orbital and lunar missions of greatly extended durations. Under these Prolonged Mission studies, the Concept II CSM is identified as a Mission Support vehicle, since it supports the laboratory (AORL) by resupplying subsystems, crew members, and expendables. The Concept III CSM is

identified as a Logistics Support vehicle since it resupplies only expendables and crewmen, with the CSM subsystems remaining in a quiescent state after docking to the laboratory module. The laboratory modules are being investigated separately (i.e., by another contractor) under a current MSC study program. The definition of the characteristics of the Apollo X laboratory modules is also included as part of this separated AORL study effort.

The primary objective of the Apollo X study has been to define a standard spacecraft design capable of alternately performing extended-duration lunar or earth-orbital missions of NASA near-term interest. Included in this primary objective were studies to determine specific modifications required for each subsystem to accomplish the extended missions. Subsystem qualification test programs, which would substantiate the analytically-derived extended-life subsystems, were also defined. In the experimental area, emphasis was placed on definition and integration of experimental packages (for earth-orbital missions) based on experimental lists provided by NASA at the initiation of the study. As an aid to NASA evaluation of the desirability of, or requirement for, the laboratory module, studies were also performed defining development factors based on CSM changes attendant with and without the laboratory module. All development factors investigations were based on an integrated, but noninterference, relationship with the current Apollo program.

The Apollo X study was initially focused toward consideration of earth-orbital experimental missions of up to 45 days duration; spacecraft and subsystems requirements were to be developed through the determination of the experimental mission requirements. Secondary studies were aimed at determining the modifications required to the 45-day earth-orbital CSM design when applied to a lunar survey mission (including up to 28 days in lunar polar orbit). Several weeks after study initiation, however, NASA directed that equal emphasis be given to the lunar mapping mission, and requested that primary emphasis be placed on the definition of a standard CSM suitable for both the lunar and earth-orbital missions. Accordingly, a reevaluation was made of previously developed earth-orbital mission requirements. This reevaluation indicated that the lunar survey mission imposed overriding requirements in most subsystem areas.

As previously indicated, the overall study approach was based on derivation of a standard multimission CSM applied to different vehicle configuration arrangements, as indicated in Figure 2.

In the basic configuration, experimental work space is made available in the command module by elimination of the third crew member and couch, and by providing for in-orbit stowage of the center couch under the pilot (left-hand) seat position. This configuration is adaptable to earth-orbital

experimental missions, but is not suitable for the lunar missions since it would be required that the service module be completely filled (with propellant, fuel cells, cryogenic tankage, etc.) with no remaining volume available in the SM for mission-oriented equipment. The basic configuration has, however, the definite advantage of being able to provide an experimental laboratory for early flights (such as biomedical and human factors experiments) without schedule dependency on a separate laboratory module. Additionally, these early flights would result in a high confidence base for performing subsequent missions with a laboratory or lunar survey mission module.

The laboratory configuration has increased volume availability, as indicated in Figure 2, through the addition of a separate laboratory module carried within the LEM adapter section. The separate laboratory module (approximately 1200-1500 cubic feet volume) is sized by the provision of space inside the LEM adapter for a second module of identical geometry. This configuration can be applied to all of the earth orbital and lunar missions of current interest since mission-peculiar payloads can be installed within the laboratory module. The configuration is extremely versatile in that the experimental volume availability can be controlled by the addition of either one or two laboratory modules. In addition, a single module can be mounted on the LEM descent stage adapter, thereby permitting additional payload for lunar missions, orbital plane change capability, or for providing a lunar landing cargo module.

All technical studies were predicated on the basis of the Apollo Block II configuration. Since past studies revealed that subsystems characteristics materially influenced configuration design, heavy emphasis was placed on subsystems definition as indicated in Figure 3. In this regard, not only primary subsystem components were examined in detail, but secondary components were also fully analyzed. In certain instances where it was found that the addition of spare or redundant Apollo subsystem components might not suffice for the extended-life application, product improvement areas were identified and tentative solutions were established.

Previous studies had identified in-flight maintenance as a major study problem area in view of the number of spares and redundant components used for subsystem life extension. Correspondingly, in-flight maintenance was examined with regard to possible methods of implementing these spares and redundancies as well as determining the overall demands on crew time. These requirements were integrated into crew and experimental station-keeping activities and served as a constraint in the experimental scheduling area. In the experiment design and packaging analyses, it was possible to examine the biomedical and behavioral measurements in great detail through the use of baseline data established in the initial Extended-Mission Apollo study. It is believed that the Apollo X biomedical and human factors experiments as now defined are sufficiently detailed to permit the establishment of equipment specifications.

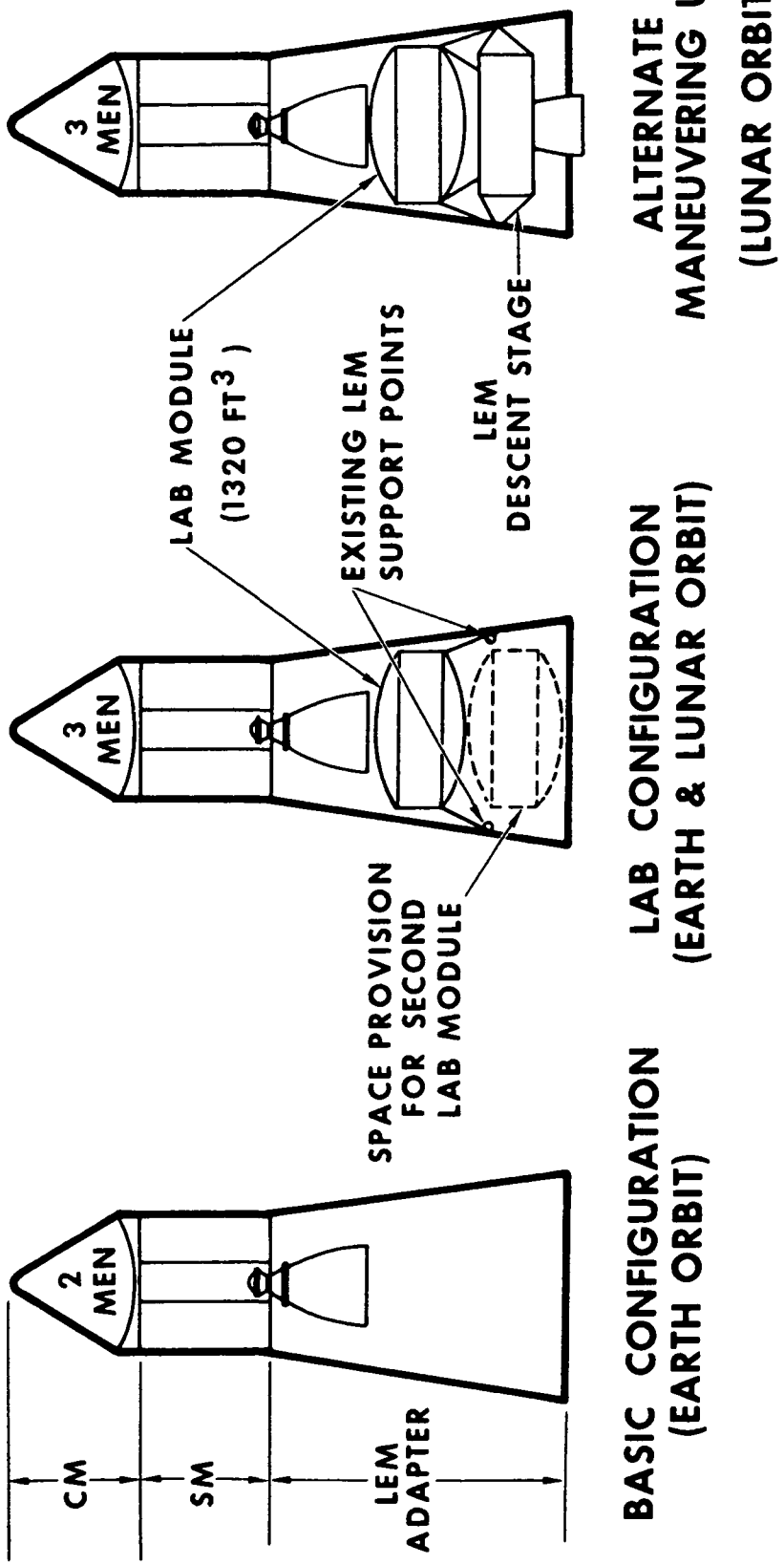

Figure 2. Apollo X Concepts

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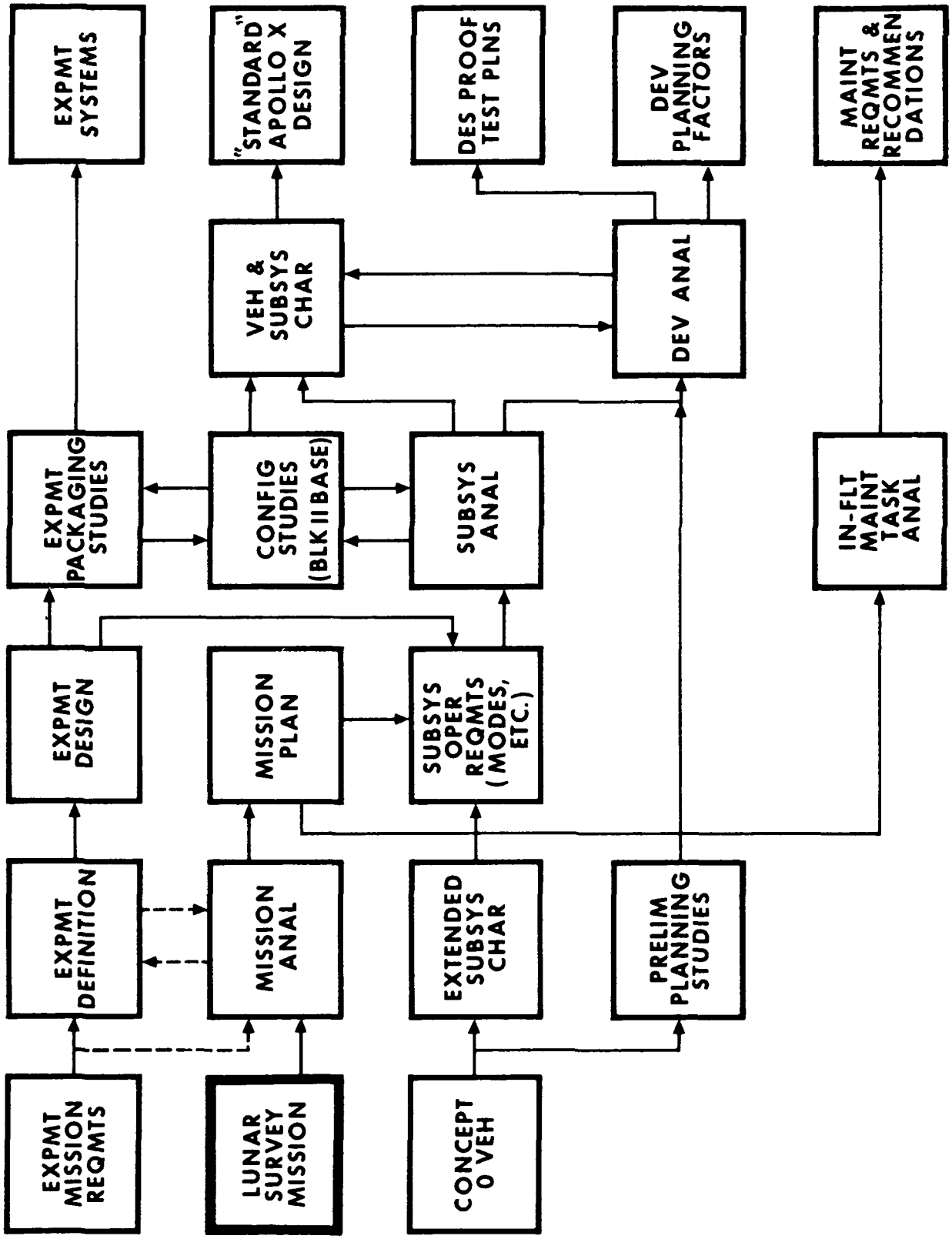

Figure 3. Apollo X Study Approach

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INTRODUCTION

The principal purpose of the Apollo X mission analysis was to develop credible spacecraft missions which could best demonstrate the ultimate life-times of Block II Apollo systems both with and without modifications. Emphasis is placed on the word credible, since there is no intention to imply that the missions, experimental packages, and equipments developed in this study are the only ones which could logically be considered in measuring the potential of Apollo systems. However, it is believed that the evolved mission concepts are fully reasonable as well as being adequate to demonstrate the potential system life capabilities.

It was decided to use the Lunar Survey Mission, a potentially desirable step toward the eventual overall lunar exploration objective, as the baseline operational requirement. Although there are several possible approaches to lunar mapping, the most stringent case is probably the 28-day lunar polar orbit mission, providing for duplicate photographic coverage (within lighting restrictions) of every point on the lunar surface. The mission as defined uses the Block II Apollo CSM, with appropriate system modifications, in conjunction with a separate laboratory, or "survey" module, carried into orbit within the LEM adapter.

Use of this mission as the configurational and performance standard provided a logical base for design of earth orbital experimental missions. If typical 45-day experimental programs could be carried out with the LSM spacecraft design, then a standard multi-mission Apollo X would be achieved. The standard Apollo X would not necessarily be optimal for any mission, operational or experimental, but would provide the greatest flexibility and be the most cost-effective.

Five versions of the earth orbital experimental mission were defined for comparative evaluation. Grouped into three basic mission plans and, in some cases, into two configurations (i.e., with or without a laboratory module), they provide a range of such factors as orbital inclination, accommodation of typical experimental packages, and subsystems demands.

Within the operational and experimental mission matrix thus developed, detailed configuration and performance analyses were completed. The objective of the performance analysis was to examine representative Apollo X mission profiles, with their concomitant environments and operating requirements, which would provide a basis for meaningful subsystem studies.

Useful payload weight characteristics for these missions were also determined to aid in the evaluation of various subsystem approaches to life extension.

To satisfy the above objectives, earth orbital missions at both low and high (polar) inclinations as well as lunar (polar orbit) missions were synthesized. The range of payload capabilities required for these missions necessitated examination of both the Saturn IB and the Saturn V launch vehicles. The lunar mission was indicated to be critical for a number of subsystems, and the abort characteristics for the lunar polar orbit mission results in significant effects on allowable payload.

For comparison of the relative costs of a multi-mission LSM standard vehicle with the costs of a spacecraft design optimized only for earth orbital experimental missions, a limited review of the major cost elements was made; both the CSM only and the CSM plus laboratory cases were evaluated. As is often true with cost-effectiveness studies at this phase of system development, the results are useful for comparative purposes (rather than absolute judgements), providing that all assumptions and qualifications are clearly identified. For choice between the laboratory module and CSM-only cases, cost effectiveness factors appear lost in the complexity of advanced space goals. However, the studies do tend to verify the hypothesis that a standard multimission spacecraft is ultimately more economical than a spacecraft designed only for earth orbital experimentation.

This volume contains mission sequence, performance, and effectiveness results of the study for each selected mission plan. Typical experimental missions are developed in Volume 5.

TECHNICAL ANALYSIS

ASSUMPTIONS AND CONSTRAINTS

In an analysis of this nature, the study assumptions and constraints must be clearly stated before technical work is initiated. These are specified for the Apollo X study as follows:

Probability of Mission Success (including 0.95 booster reliability)	At least 0.90 for all missions
Probability of crew safety	At least 0.999 for all missions
Launch vehicle	
Low inclination (285°) Earth orbit	SI-B
Polar Earth orbits	S-V
Lunar orbit	S-V
Launch Site	Cape Kennedy for all missions
Orbit Altitude	
All Earth orbits	Nominal 200 n.m.
Lunar orbit	Nominal 80 n.m.
Crew Size	2 men for all CSM configurations 3 men for all CSM/lab configurations
Normal Recovery Zones	
Low inclination Earth orbit	Atlantic Ocean near Florida
Polar Earth orbit	Atlantic Ocean near Florida
Lunar orbit	LOR Apollo sites (Bora Bora and Hawaii)

Note: All normal landings will be in daylight

Contingency Recovery Zones	Water sites near Okinawa, Pearl Harbor, and Florida
Low inclination Earth orbit	
Polar Earth orbit	Water recovery near 45° S. latitude, (either day or night)

GOSS Communications Stations	All current and planned communication stations to be in operation (including unified S-band and DSIF stations)
External Geometry	There will be no deviation from Block II Apollo geometry during the launch and ascent phases.

MISSIONS AND CONFIGURATIONS

Immediate spacecraft derivatives of Apollo in all probability will be restricted to near-earth and lunar missions. For obvious economic reasons, it is desirable to design an Apollo X that can be used for either type mission with relatively minor modifications.

Various lunar missions can be postulated for Apollo X in the 1967-72 time period. These missions include photographic and spectrozonal mapping of the entire lunar surface, detailed reconnaissance of areas of interest for lunar landing sites, and operational and logistics support for lunar bases. The lunar mapping mission is selected for comparative design purposes for several reasons. First and most important, this mission imposes the greatest life extension and reliability demands on the stabilization and control, reaction control, and guidance and navigation systems because of the rigid attitude and navigational control requirements associated with mapping. Secondly, the long total time spent in a local vertical orientation implies a worst-case thermal control condition. Finally, the lunar polar orbit associated with the mapping mission necessitates a greater ΔV capability for a free return to earth. Thus, the lunar mapping mission is ideal for demonstrating the ability of Apollo to perform a variety of lunar missions because of its near maximum demands on no less than five key Apollo systems (i.e., stabilization and control, guidance and navigation, reaction control, environmental control, service module propulsion.).

The earth-orbit missions imply equal or lesser system demands than lunar missions of corresponding nature and duration for the following two reasons:

1. Much less service module propellant is required for either a low inclination or a polar earth orbit than for any lunar orbit.

2. Lower reliability in many command and service module systems can be tolerated in earth orbit (for the same crew safety and mission success probabilities) because of the much shorter return-to-earth time.

Therefore, the 28-day lunar mapping configuration serves as an excellent design goal for an Apollo vehicle that is feasible for a wide spectrum of either near-earth or lunar missions.

It is desirable to plan a sequence of missions such that Block II Apollo is modified in an orderly fashion for increasingly complex missions. It appears that a good way to accomplish this plan is to schedule, if possible, all two-man missions before any three-man laboratory configurations are flown. The major changes to the Block II Apollo for its conversion to a two-man, 45-day earth orbiting space station are listed below. (The rationale for these changes is described in the Trade-Off Analysis Section of this volume.)

Add two-gas atmosphere.

Replace three-man couch system with two-man system.

Add LiOH canisters to the command module.

Add crew supplies (90 man-day total).

Add three fuel cells in the service module.

Add reactant storage (45-day total) in the service module.

Add ECS cryogenic storage (135 man-day total) in the service module.

Replace the service module propulsion tanks with small tanks.

Remove the G&N system (except optics).

Add spares and redundancies to meet 45-day earth mission reliability goals.

By the time the two-man flight series has been completed, the laboratory module should be available for the more complex three-man earth missions. The Block II crew couch and G&N systems can be readily restored. The lithium hydroxide canisters in the command module can be eliminated, because extra Apollo-type canisters can be stored in the laboratory and brought into the command module as needed. The service module propulsion tanks remain the same size as in the two-man vehicle for low inclination orbits. (But these must be increased to one-half Block II size if earth polar orbit missions are flown.) The cryogenic tankage in the service module will not be changed from the two-man vehicle, because the tanks in that configuration are sized for three-man 45-day missions. The most important system

change is an increase in the size of the service module reaction control system tankage, which results from substituting LEM RCS tankage for Block II RCS fuel and oxidizer tanks. These changes enable the Apollo X to meet the orientation requirements of the earth mapping mission.

The lunar mapping mission will be a logical follow-on to the series of three-man earth orbit missions. The crew functions and operation of the spacecraft should have become fairly routine, and enough system lifetime and reliability data will have been accumulated to establish sparing and redundancy levels for the desired mission success and crew safety probabilities. The Block II service module propulsion tankage is restored to the service module, and horizon sensors are added in the command module to reduce G&N operating time during mapping.

It is important to note that the configuration of the service module is the key to smooth application of the plans previously mentioned. There are two service module systems that are associated with all three mission types. These systems are the five fuel cells with reactant tankage for 45 days and the ECS cryogenic tankage for 135 man-days. Thus, development and testing costs can be substantially reduced if the designs and locations of these two systems are not changed between any of the Apollo X configurations. With these systems fixed, spherical service module propulsion tanks must be located in the service module for the two-man configuration, while space must be found in the service module for the LEM RCS tanks and the large service module propulsion tanks required for the lunar mapping configuration. Hence, it is clear that the lunar mapping mission imposes the most stringent demands on the service module volume and system arrangements as well as on system performance as previously described. The validity of the lunar mapping mission as a goal for Apollo X configuration and system design is, therefore, confirmed.

CONCEPTUAL DEFINITION OF AN APOLLO X 28-DAY LUNAR MAPPING MISSION

To facilitate synthesis of a lunar mapping Apollo X configuration, an outline of one possible conceptual lunar mapping mission has been prepared. This mission is planned to conform with present NASA and scientific thinking so that it can be performed by a Block II Apollo after certain feasible modifications. The major mission factors considered are objectives, trajectory, and ΔV requirements, lighting, vehicle configurations and weights, mapping equipment, mapping operations, attitude control, and thermal control.

MISSION OBJECTIVES

The purpose of the mission is to conduct a manned flight in orbit around the moon for 28 days, during which time the entire surface of the moon is photographed with high resolution cartographic and spectrozonal cameras. Thus, accurate flat and relief maps of the moon could be made for use in future lunar operations.

MISSION PROFILE

The Apollo X will be launched by the Saturn V from Cape Kennedy when the earth-moon position is favorable. After burning for approximately 2-1/2 minutes, the S-IC stage shuts down and separates. About 3 seconds later, the S-II stage ignites, and the LES tower is jettisoned. The S-II burns for approximately 6 minutes and then separates. The S-IVB stage is then ignited and burns for about 3 minutes until a circular orbit of 100 nautical miles is attained. After a nominal time of several hours in parking orbit, the S-IVB is ignited for the second time and fires for about 5 minutes to inject the spacecraft into translunar flight.

The laboratory transposition and docking maneuvers are completed between 30 and 60 minutes after lunar injection. After transposition, the S-IVB stage is separated. The first midcourse correction is made about 12 hours after launch with a second correction made as the spacecraft enters the lunar sphere of influence. A final correction is completed about one hour before the end of the translunar flight. Approximately 72 hours after launch, the service module engine is retrofired to attain circular polar lunar orbit of 80 nautical miles.

After completion of lunar operations, the command and service modules are separated from the laboratory, assume attitude for lunar escape, and are injected into a lunar escape orbit. At a point near the lunar sphere of influence, about 18 hours after escape, a relatively large midcourse correction is applied to establish the proper transearth injection conditions. Second and final course corrections are made later as the Apollo X approaches the earth. Approximately 4-1/2 days after leaving the vicinity of the moon, earth entry occurs at 400,000 feet. At 25,000 feet, the drogue parachute is deployed; and at about 15,000 feet, the main parachute is deployed. Intended landing sites are the lunar orbital rendezvous Apollo sites near Bora Bora or Hawaii.

MANEUVERING REQUIREMENTS

The total service module engine ΔV requirements with no main propellant off loading are approximately 9800 feet per second. This total

consists of 300 feet per second for translunar midcourse corrections, 3000 feet per second for retro into an 80-nautical-mile circular polar lunar orbit, and 6500 feet per second for escape from a lunar polar orbit (free return) and transearth midcourse corrections.

LIGHTING REQUIREMENTS

Because photographic conditions for mapping are best when there are some shadows to heighten surface definition but when the shadows are not so long that they obscure surface details, the polar lunar orbit is selected accordingly. The best evidence to date indicates that the optimum region for mapping lies between 45 degrees and 75 degrees from the subsolar point. It can be readily shown that, if the spacecraft is injected into a polar orbit that intersects the lunar equator 75 degrees east of the subsolar point at the start of the mission, all of the lunar surface area located between 75 degrees north and 75 degrees south selenographic latitude can be mapped in 28 days under the optimum lighting conditions. (This is true because the sun angle changes about 28 degrees during a 28-day mission.) Although the lunar polar regions can never be brought into the best lighting zone and are thus photographed under somewhat less favorable conditions, this problem is compensated for, at least in part, by the great amount of duplicate coverage at the poles.

VEHICLE CONFIGURATIONS AND WEIGHTS

The three-man configuration with a laboratory consists of modified Apollo (Block II) command and service modules docked to a laboratory module of about 1300 cubic feet that contains the cameras and associated equipment. The major modifications to the Block II Apollo would include the following:

Two-gas atmosphere

Increased ECS cryogenics and tankage for longer mission

Additional LiOH in laboratory for longer mission

Two additional fuel cells in the service module

Increased fuel cell reactants and tankage in the service module for longer mission

Increased RCS propellant and tankage

Redundant G&N components for increased reliability

Possible active thermal control for service module RCS thrusters.

Modifications to G&N and SCS electronics and logic for precise attitude control with respect to local vertical possibly including a horizon sensor

A typical weight summary is as follows:

Apollo X command module		9,660
Apollo X service module		13,785
Service module RCS propellant		830
Service module engine propellant		39,000
Laboratory structure		1,660
Cameras, film and film containers		
Cartographic camera	100	
Spectrozonal camera	90	
Television camera & display unit	135	
Film (returned in command module)	80	
Film cassettes (60 pounds returned in command module)	185	590
Total weight at liftoff		65,525 pounds

The service module main engine propellant can be offloaded to provide additional laboratory equipment payload if desired. However, the 9800-feet-per-second ΔV minimum requirement must be met in all cases.

MAPPING EQUIPMENT

The basic mapping equipment package consists of a cartographic camera for photography, a spectrozonal camera for analysis of the lunar surface, a television camera and display unit, and film cassettes. The film is developed after return to earth. The specifications of typical cameras are:

Specifications	Television	Cartographic	Spectrozonal
Focal length	Variable	6.2 inches	6.2 inches
Resolution	1000 TV lines per inch	45 lines/mm	60 lines/mm

Specifications	Television	Cartographic	Spectrozonal
Film speed	Not applicable	ASA 64	ASA 110
Image format	0.44 x 0.44 inch	9 x 9 inches	2-1/4 inches x 2-1/4 inches
Field of view	Variable	72 x 72 degrees	72 degrees x 72 degrees
Weight of camera	45 pounds	100 pounds	90 pounds
Feet of film	Not applicable	3200 feet	600 feet
Weight of film	Not applicable	75	5
Weight of film cassettes	Not applicable	160	25
Spectrographic range	Not applicable	Not applicable	4000 to 9000 Angstroms

MAPPING OPERATIONS

A point on the moon's surface directly under the orbital plane of the spacecraft rotates 180 degrees relative to this orbit plane in slightly more than 13.75 days. Because mapping is done only in direct sunlight, the total required mission length is about 28 days. The orbital period of the Apollo X in an 80-nautical-mile orbit is almost exactly 2 hours. In this time period, the moon has turned 1.07 degrees of selenographic longitude under the orbit plane. This turning is equivalent to a 17.5-nautical-mile-wide strip of the moon's surface at the lunar equator.

The 72 degree field of view of the cartographic camera allows a picture to be taken of a strip about 116 nautical miles wide, which, allowing for a 25 percent overlap, reduces to a strip approximately 87 nautical miles wide. Thus the camera could be operated only on the first, fifth, tenth, etc., orbits and still adequately photograph the whole surface at the lunar equator. However, camera operation during the entire solar illuminated period on every fifth orbit will result in excessive duplication at the polar regions. Adequate overlap is obtained if camera operation is confined only to the areas between 45 degrees north and 45 degrees south selenographic latitudes on every fifth orbit and between pole and pole on every tenth orbit.

There are approximately 330 orbits in the 28-day mission. Because the orbit period is about 2 hours, the mapping scheme previously described requires a total camera operating time of

$$\frac{330(1)}{10} + \frac{330}{10}(1/2) = 49.5 \text{ hours}$$

Because the circumference of the moon is 5900 nautical miles, the spacecraft overflys a frame area (87 nautical miles along the orbit trace)

every 1.83 minutes. Thus the total number of pictures taken is approximately (49.5 x 60)/1.83 = 1623. This number of pictures represents 1623 (9/12) = 1217 feet of 9-inch film.

For the above mapping plan the operational profile is as follows:

Operation	Elapsed Time		Cumulative Time	
	Hours	Minutes	Hours	Minutes
Start at pole	00	00	00	00
Mapping interval	01	00	01	00
No mapping	09	15	10	15
Mapping interval	00	30	10	45
No mapping	09	15	20	00
Mapping interval	01	00	21	00
No mapping	09	15	30	15

Cycle to 28 days

When a duplicate photograph is required of every point on the lunar surface for greatest stereoptical effect, the cameras must be operated more frequently. Operation from 45 degrees north to 45 degrees south selenographic latitude on every third orbit and from pole to pole on every sixth orbit provides the necessary coverage. The total camera operating time under these conditions is

$$(\frac{330}{6})\,(1) + \frac{330}{6}\,(1/2) = 82.5 \text{ hours}$$

The total number of pictures taken is

$$\frac{82.5 \times 60}{1.83} = 2705, \text{ representing 2029 feet of 9-inch film.}$$

The operational profile under this alternative is as follows:

Operation	Elapsed Time		Cumulative Time	
	Hours	Minutes	Hours	Minutes
Start at pole	00	00	00	00
Mapping interval	01	00	01	00
No mapping	05	15	06	15
Mapping interval	00	30	06	45
No mapping	05	15	12	00

Operation	Elapsed Time		Cumulative Time	
	Hours	Minutes	Hours	Minutes
Mapping interval	01	00	13	00
No mapping	05	15	18	15

Cycle to 28 days

ATTITUDE CONTROL

Yaw or roll maneuvers not in the direction of orbital motion are avoided during mapping, if possible, because they are likely to complicate the problems of alignment and of matching overlapping photographs. To reduce the number of RCS starts and propellant, the vehicle could be given a slow motion in the pitch direction at the end of each mapping interval of such a magnitude that the vehicle would tend to maintain a local vertical orientation until the next operating period when vernier attitude adjustments would again be made.

During mapping, the navigation and attitude requirements which may necessitate SCS/RCS and G&N modifications are as follows:

Orbit velocity measured to 4.0 feet per second

Orbit altitude measured to 1500 feet

Drift rate no more than 0.02 degrees per second about axes not held to local vertical

Attitude with respect to local vertical held to ±0.50 degree

THERMAL CONTROL

The relation between mapping operations and thermal control methods is one of the most important factors to be considered in planning a 28-day lunar mapping mission and thus merits a detailed analysis.

As previously noted, the optimum region for mapping lies between 45 degrees and 75 degrees from the subsolar point. Furthermore, if the angle between the orbit plane and the subsolar point is 75 degrees at the start of the 28-day mission, this angle is reduced to approximately 47 degrees at the end of the mission. This means that the worst case from a thermal standpoint occurs at the start of the mission when the sun's rays strike one side of the vehicle at an angle 15 degrees from the

vertical for an uninterrupted period of up to one hour, while the opposite side of the vehicle is exposed to cold of deep space. Some reports indicate that an angle as small as 10 degrees from the subsolar point is marginally satisfactory for cartographic mapping (although not for low-altitude high-resolution photography). If the orbit plane makes an angle of 38 degrees from the subpolar point at the start of the 28-day mission, this angle will not be reduced below 10 degrees by the end of the mapping operations. In this case, at the start of the mission the sun's rays strike the vehicle at an angle of 52 degrees from the vertical. Assuming the Apollo X (command module, service module, and laboratory) with passive thermal control cannot satisfactorily handle the heat loads and temperatures implied by the optimum photographic conditions, it must be determined whether there is some starting angle between 75 degrees and 38 degrees that is feasible for passive thermal control. If a thermal computer program establishes this feasibility, then the Apollo X can utilize passive thermal control and can limit its maneuvers during mapping to pitch changes for local vertical hold over a maximum period of one hour.

The next situation to consider is where heat loads and/or temperatures are unacceptable for all angles (between the subsolar point and the orbit plane) greater than 38 degrees. In this case there would be five possible solutions, four of which embody the passive thermal control concept. Each of these possibilities is analyzed in the following paragraphs that list method of spacecraft operation, advantages, disadvantages, and the data required for tradeoffs to arrive at the best solution.

Possible Solution I

Operation

The Apollo X is given a slow yaw motion about its Y axis at the same time that the Y axis is held to an approximate local vertical. The camera is on a three-axis independent platform which is oriented during mapping operations to a precise local vertical.

Advantages

Passive thermal control is effectively obtained with moderate RCS fuel expenditure.

Disadvantages

An expensive and complex three-axis independent camera mounting platform is required which is otherwise unnecessary if Apollo stabilization and control system can attain desired attitude hold accuracy.

Data for Tradeoffs

Ability of Apollo X S&C system to meet photographic attitude hold tolerances and implied RCS fuel requirements for 82.5 hours of mapping time.

Additional RCS fuel requirements for slow yaw about Y axis for 82.5 hours mapping time.

Approximate weight, cost, and power drain of three-axis camera platform and associated electronics, stable platform, horizon sensors, etc.

Possible Solution II

Operation

The Apollo X is given a slow roll about its X axis at the same time that the X axis is held to an approximate local vertical. The camera is on a two-axis independent platform that is oriented during mapping operations to a precise local vertical. The camera is mounted along the centerline of the laboratory module and points along the +X axis.

Advantages

Passive thermal control is effectively obtained with a minimum of RCS fuel. The moment of inertia is least about the X axis, and advantage is taken of the small lunar gravity gradient. The camera and its platform are conveniently located at outboard end of the laboratory module.

Disadvantages

An expensive and complex two-axis independent camera mounting platform is required, which is otherwise unnecessary if Apollo stabilization and control system can attain desired attitude hold accuracy. The addition of a second laboratory or logistics module would prevent the use of the camera.

Data for Tradeoffs

Ability of Apollo X S&C system to meet photographic attitude hold tolerances and implied fuel requirements for 82.5 hours of mapping time.

Additional RCS fuel requirements for slow roll about the X axis for 82.5 hours mapping time.

Approximate weight, cost and power drain of a two-axis camera platform, and associated electronics, stable platform, horizon sensors, etc.

Possible Solution III

Operation

The Y axis of the Apollo X is held to the local vertical within required photographic tolerances during the instant at which the film is actually being exposed. An exposure will occur approximately every 1.83 minutes of the 82.5-hour total mapping interval. After each exposure, the vehicle is given a 360 degree roll along the X axis and restabilized for the next exposure. The camera does not have an independent mount.

Advantages

No separate camera platform is required by reason of thermal control maneuvers. Passive thermal control is obtained.

Disadvantages

The RCS fuel requirements are very great. Rough calculations show that fuel consumption would be from 1.5 to 3 pounds per maneuver for the roll maneuver alone. Additional fuel would be required for restabilization. Because there are about 33 exposures per hour or a total of over 2700 during the entire mission, this method is obviously infeasible on a weight basis.

Possible Solution IV

Operation

The Y axis of the Apollo X is held to the local vertical within required photographic tolerances during the mapping operations. A thermal computer program is used to establish the maximum feasible continuous time interval for mapping (from a thermal load and temperature standpoint) with the lunar orbit established for optimum photographic lighting (75 degrees from the subsolar point at start of mission). Mapping is then done on each orbit for the latter time interval. The latitudes covered in successive orbits are staggered. For example, with a 10-minute maximum permissible local vertical hold, the vertical might map from 90 degrees north latitude to 75 degrees north latitude on the first orbit, from 75 degrees north latitude to 60 degrees north latitude on the second orbit and so on, with cycling every sixth orbit.

Advantages

No separate camera platform is required by reason of thermal control maneuvers.

Optimum illumination for photography is possible. No additional maneuvers are required for thermal control during the mapping operations, and passive thermal control is obtained.

Disadvantages

The images form an irregular pattern and may be more difficult to integrate. Mapping on every orbit is much less satisfactory from a standpoint of human factors. The increased number of mapping intervals implies a significant increase in the total operating time of critical G&N and stabilization and control systems (for warmup and realignment purposes).

Data for Tradeoffs

A new operational profile employing the staggering technique.

Ability of Apollo S&C system to meet photographic attitude hold tolerances and implied fuel requirements for the staggered operational profile.

Evaluation of image integration problems.

Reliability penalties because of increased number of startups.

Possible Solution V

Operation

The Apollo X is equipped with the Block I active thermal control system.

Advantages

Attitude maneuvers are required only for maintenance of a local vertical orientation during mapping.

Optimum illumination for photography is possible. No separate camera platform is required because of thermal control maneuvers.

Disadvantages

The power drain of the RCS heaters may be significant for long mapping times.

Data for Tradeoffs

Weight, cost, and power drain of an active thermal control system.

Ability of Apollo X S&C system to meet photographic attitude hold tolerances and implied fuel requirements for 82.5 hours of mapping time.

SELECTION OF REPRESENTATIVE EARTH ORBIT EXPERIMENTAL MISSIONS

A more detailed description of the individual experiments for each mission is given in Volume 3 of this report. Only the general philosophy and rationale for the choices will be set forth in this volume.

The chief principles governing the selection of the missions are summarized as follows:

Each mission must rank highly (during the Apollo X time period) on a scale of product value and/or relevancy to the national space program, using the following effectiveness criteria:

Number of experiments per mission

Number of experiment hours per man per mission

Nature or type of specific experiments included per mission

Variety of experiments included per mission

No mission can result in any significant hazard to the crew safety. To this end, the basic safety package of critical biomedical and behavioral measures is incorporated in all missions and certain hazardous experiments are excluded.

The missions must be associated with a large number of variations in system demands, flight profiles, and configurations to demonstrate most effectively the extension limits of Apollo systems.

The time-sequencing of the missions must conform with the expected availability of the required configurations, with a feasible progression of modifications to Block II systems, and with the published goals of the national space program.

MISSION PLAN 1

Two-Man Biomedical and Behavioral Experiments – Mission 1A

A complete knowledge of the long time effects of zero g on the crewmen obviously has the highest priority among all possible missions. This knowledge of zero-g effects must be obtained by conducting certain biomedical and behavioral tests on the crew while they are in the weightless state. It is also safe to assume that an Apollo X without a laboratory module will be available for use at an earlier time period than will be any vehicle with a lab module.

Thus, the first mission selected is one that utilizes a two-man Apollo X command and service module to conduct a large part of the biomedical and behavioral tests. Limited extravehicular activities will be conducted to demonstrate the feasibility of crew activities in space. Some engineering and operational experiments (as crew time allows), will complete the mission.

The recovery, communication, and cost advantages of a low inclination orbit and the fact that the weightless-effects test have no dependence on the orbit inclination dictate the selection of a 28.5-degree orbit for this mission.

Three-Man Biomedical, Behavioral, Engineering, and Scientific Experiments – Mission 1B

While it is shown in this study that a great many of the biomedical and behavioral tests can be performed in the Apollo X command module, the additional work space made available by incorporating a laboratory module permits the inclusion of nearly all of the tests not requiring an attending physician. The laboratory also adds a significant (although quantitatively unknown) capability for more efficient performance of these tests. In addition, the use of a laboratory module in Mission 1B allows a variety of valuable scientific, operational, and engineering experiments to be performed without compromising the biomedical and behavioral experiments. These scientific and engineering tests will establish performance parameters for systems, hardware, and operations that must be obtained prior to extended space missions, and thus they have priority next highest to the biomedical and behavior experiments. An early extensive evaluation of man's extravehicular activities in space is a goal of the manned space program. The probable time period for Mission 1B allows time for the development of an air lock on the laboratory and thus permits convenient outside egress. Because none of the included experiments are orbit sensitive, a 28.5-degree orbit is selected for the same reasons as in Mission 1.

MISSION PLAN 2

Three-Man Engineering and Scientific Experiments — Mission 2A

This mission is primarily devoted to the performance of a large number of natural science, engineering, and operational experiments that cannot be scheduled on earlier missions because of weight, volume, or crew time limitations. Some of these experiments involve the use of external sensors, booms, and pods that are fitted to the aft end of the laboratory module. The experiments are selected not only because they provide valuable data but also because they are a means to measure the performance limits of both crew and hardware. For example, the extravehicular activities required in connection with some of the experiments have the effect of measuring crew aptitude for this activity and, additionally, of disclosing tool and equipment requirements. A safety package of the biomedical and behavioral measurements, considered minimal for crew safety checks, completes the mission A 28.5-degree orbit is best for this mission.

Three-Man Astronomical Scientific and Engineering Experiments - Mission 2B

Experiments utilizing a telescope for solar and space observations above the disturbing effects of the terrestrial environment may be of great value, and they merit inclusion on this mission. The size of the telescope and its mount necessitates a vehicle having a laboratory module. As a result, there is volume available for conducting a number of other scientific and engineering experiments, which can extend the knowledge gained in Mission 2A. The same safety package as in Mission 2A is included. A 28.5-degree orbit is satisfactory for this mission.

MISSION PLAN 3

Three-Man Mapping — Mission 3

An earth mapping mission imposes unique, and in some cases maximum, demands on both space vehicle crew and systems. A listing of Apollo X missions ordered for difficulty would, therefore, place Mission 3 after the missions previously described. Since, however, the earth mapping mission is the only earth mission that is strictly comparable to a lunar mapping mission, an urgent need for the latter might cause a change in the order of the missions. Regardless of the actual sequence of missions, the earth mapping operations are essential in order to confirm Apollo X performance capabilities for lunar mapping, particularly in regard to the environmental control,

guidance and navigation, and stabilization and control systems Techniques for geodetic control in photogrammetric mapping from a space vehicle should also be developed and checked during Mission 3 before the more difficult lunar mapping is attempted. Under assumptions made earlier about the lunar survey mission plan to be applied, a laboratory module is required for adequate performance of the mission. This allows the inclusion of some scientific, engineering, and biomedical experiments as well as the safety package. An earth polar orbit is preferred in order to approach more closely the propulsion and weight constraints of lunar mapping and in order to permit operational mapping of the earth. Nevertheless, if a low inclination orbit is mandatory because of launch site constraints, the primary function of testing the mapping equipment and operations can be performed quite adequately

MISSION-CONFIGURATION RELATIONSHIPS

The relationship of the missions to the previously described configurational changes is shown in Table 1.

Table 1. Apollo X Probable Mission and Configuration Sequence

Time Period	Mission	Configuration
1968	Apollo near-earth and lunar reconnaissance missions	Block II Apollo
Early (1968)	Biomedical and behavioral experiments with two-man crew (Apollo X Mission 1A); some engineering experiments	Add two-gas atmosphere Replace three-man couch system with two-man couch system Add crew supplies (90 man-day total) Add two fuel cells in SM Add reactant storage in SM (45 day total) Add ECS cryogenic storage in SM (135 man-day total) Resize SPS tankage for earth orbit Remove G&N system (except optics) Add spares and redundant components to meet 45-day earth mission reliability goals
Intermediate (1969)	Biomedical and behavioral experiments with three-man crew, scientific, engineering, and operational experiments, extravehicular activity (Mission 1B) Scientific, engineering, and operational experiments, extravehicular activity (Mission 2A)	Add laboratory module Restore three-man couch system Add crew supplies (135 man-day total) No additional modifications required

Table 1. Apollo X Probable Mission and Configuration Sequence (Cont)

Time Period	Mission	Configuration
Intermediate (1969) (Cont)	Earth mapping, biomedical, scientific, and engineering experiments (Mission 3)	Add horizon sensors; replace RCS tankage with LEM RCS tankage; restore G&N system; add active thermal control for RCS thrustors
Late (1970)	Lunar mapping	Add spares and redundant components for lunar mission reliability. Restore Block II SPS tankage
	Astronomical observation, scientific, engineering, and operational experiments extravehicular activity (Mission 2B)	Resize SPS tankage for earth orbit

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MISSION LIMITATIONS

It is appropriate here to point out some of the limitations of the preceding selection of missions in determining ultimate Apollo X system capabilities.

In the first place, it is unavoidable for reasons of safety that the early missions be planned primarily around the biomedical and human factors experiments. However, only minor attention has been given to the effects on the vehicle or systems of a requirement for artificial-g which might become known as the result of these experiments. This requirement would most likely be met by a centrifuge that could have a significant effect on the electrical power and stabilization and control systems of the Apollo X, as well as on the structural design and arrangement of the lab module. The implications of artificial-g devices on vehicle design will have to be thoroughly studied before actual configurations are defined.

Secondly, extravehicular activities have not been considered in a true operational sense; i.e., with the aid of specially designed tools, power backpacks, and the like. It is evident that a more realistic evaluation of EVA should be made in subsequent studies to determine the implications on Apollo X design if the crew has ready access to the service module via the space environment.

Another area in which performance capabilities are not fully tested is the power system. Most of the selected experiments draw very little power; consequently, the main concern has been with the life of the fuel cells rather than with available power output. In the future, it may be desirable to include equipment in the Apollo which would have a relatively large power requirement. Problems that could arise from such a power loading should be fully investigated.

Finally, the system demands imposed by extensive maneuvers during orbit have not been assessed. These include plane changes as well as radial changes which can result in a very high ΔV and fuel requirements, particularly in the case of plane changes. Nonmilitary missions have not yet been postulated which require such maneuvers; however, it must be emphasized that systems and tankage in the service module would be radically effected by a need for a significant space maneuver capability.

DETAILED SCHEDULING OF MISSION PACKAGES

The experiment packages in Missions 1A, 1B, and 2B have been defined and are used in the computer program for establishing experiment-time schedules that are consistent with system and crew performance capabilities. The cases are handled in the order given, because this sequence is associated with a steadily increasing level of program subroutine utilization. The detailed results of the scheduling process are reported in Volume V of this report.

MISSION SEQUENCES AND SYSTEM TIMELINES

In order to determine system-life requirements and to supply certain inputs to the computer scheduling program, it is necessary to prepare event sequences and systems timelines for missions, both with and without a laboratory module, in 28.5-degree and 90-degree earth orbit and in lunar polar orbit. System operating times resulting solely from experiments are not included in these lists, because the specific crew time and system demands implied by each experiment are assembled in a separate input to the computer program. However, for a given mission, the total length of operating time for each system can be found by adding the experimental requirements to the nominal requirements shown in the system timelines.

The Apollo X-with-laboratory configuration in a low (28.5-degree) inclination orbit is taken as the standard case. For simplicity, the remaining three earth-orbital missions are treated as variations from the standard.

EARTH MISSION, CSM WITH LABORATORY MODULE, LOW-INCLINATION ORBIT

Launch Boost Phase Event Sequence (10 Minutes 23 Seconds Duration)

The event sequence for the launch boost phase is given below. Table 2 gives system operation times.

1. The S-1 propulsion system burns for 2 minutes 33 seconds, followed by a 3-second staging delay before S-IVB stage ignition. The S-IVB stage burns for 7 minutes 47 seconds to inject the spacecraft into an 80-nautical-mile-altitude parking orbit.

2. The environmental control system is put into operation 2 hours before launch and remains in operation thereafter.

3. The communications system is put on standby 2 hours before launch following prelaunch checkout.

4. The guidance and navigation system is put on standby 2 hours before launch following prelaunch checkout.

5. The stabilization and control system is put on standby 2 hours before launch following prelaunch checkout.

6. The electrical power system is activated 4 hours before launch and remains in operation thereafter.

7. The launch vehicle emergency detection system is activated 30 seconds before launch and is used only during the first stage operations.

8. The launch escape system is operating during the initial portion of the boost phase. The tower is jettisoned 2 minutes 49 seconds following launch.

Table 2. System Operations, Launch Boost Phase

System	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control	-2						2	10	23	2	10	23
Communication	-2						2	10	23	2	10	23
Guidance and navigation (standby)	-2						2	10	23	2	10	23
Electrical power	-4						4	10	23	4	10	23
Stabilization and control (standby)	-2						2	10	23	2	10	23
Launch vehicle emergency detection			30		2	33		3	03		3	03
Launch escape					2	49		2	49		2	49
Propulsion												
S-1					2	33		2	33		2	33
S-IVB		2	36		10	23		7	47		7	47
Mission cumulative time									10 minutes 23 seconds			

Orbit Attainment Phase Event Sequence (2 Hours 14 Minutes 47 Seconds Duration)

The event sequence for the orbit attainment phase is given below. Table 3 gives system operation times.

1. Following S-IVB shutdown, a 9-minute-47-second period is used for a safety check and for preparations for CSM transposition and dock with the laboratory module. GOSS is notified of readiness, and the squib charges for the skirt fairing structure are activated. Approximately 15 minutes are required for the transposition and docking operations. The S-IVB is then separated, and the vehicle is positioned in the correct attitude for transfer to the final orbit by the service module propulsion. After ejection, there is a 44-minute-50-second coast period to 200 nautical miles altitude where the service module engine again fires to circularize the orbit.

2. The environmental control system is in operation during the entire phase.

3. Communications are used frequently throughout the phase.

4 The guidance and navigation system is used to obtain required vehicle attitude prior to the service module firings. The system is on standby during the remainder of this phase.

5. The electrical power system is operating during this phase.

6. The stabilization and control system is in use during the transposition and docking period, during attitude alignment prior to ejection from the parking orbit, and prior to final orbit circularization. The system is on standby during the remainder of this phase.

7. The propulsion system of the service module is used for 10 seconds to eject the vehicle from the 80-nautical-mile parking orbit and for 10 seconds to circularize the orbit at 200 nautical miles.

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Table 3. System Operations, Orbit Attainment Phase

System	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control			-			-	2	14	47	4	25	10
Communications			-			-	2	14	47	4	25	10
Guidance and navigation			-			-	2	14	47	4	25	10
Electrical power			-			-	2	14	47	6	25	10
Stabilization and control: Standby			-			-	1	49	47			
Transposition, docking, and S-IVB separation		20			35			15				
Attitude alignment prior to first service module engine firing	1	35		1	40			5		4	25	
Attitude alignment prior to second service module engine firing	2	20		2	25			5				
Skirt structure jettison		20			20	1			1			
Propulsion - service module engine												
First burn	1	40		1	40	10			10		20	
Second burn	2	25		2	25	10			10			
Mission cumulative time	2 hours 25 minutes 10 seconds											

Initial Orbital Operations Event Sequence (1 Hour 55 Minutes 34 Seconds Duration)

The event sequence for initial orbital operations is given below. Table 4 gives system operation times.

1. Upon attainment of circular orbit, communications are established with GOSS, and the orbital parameters are verified. The crew conducts a 40-minute safety check to confirm the functional performance of all systems.

2. Small orbital corrections are made if necessary, and the vehicle is positioned in the desired attitude and given a slow motion for passive thermal control.

3. The environmental control system is in operation during this phase.

4. The communication system is used for contacts with GOSS. Upon completion of system check-out and orbital adjustments, GOSS is informed that the experimental mission has begun.

5. The guidance and navigation system is used to determine the attitude of the vehicle before making final small orbital corrections. After verification that the desired orbital parameters have been attained, the guidance and navigation system is inactivated until the twenty-third day of the mission

6. The electrical power system is operating during this phase.

7. The stabilization and control system is on standby until used for vehicle alignment during the final, small, orbit corrections.

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Table 4. System Operations, Initial Orbital Operations

System	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		1	55	34	6	20	44
Communications		-			-		1	55	34	6	20	44
Guidance and navigation	2	25	10	3	55	10	1	30	00	5	55	10
Electrical power		-			-		1	55	34	8	20	44
Stabilization and control: standby	2	25	10	3				34	50			
Orbit corrections	3			3	5			5		5	4	50
Mission cumulative time	4 hours 20 minutes 44 seconds											

Orbital Operations Event Sequence - First Day Through Twenty Second Day (528 Hours Duration)

The event sequence for orbital operations (first day through twenty-second day) is given below. Table 5 gives system operation times.

1. Upon achieving the desired orbit, the crew begins the routine which is followed for the first 22 days of the mission.

2. The environmental control system is in continuous operation during this phase

3 The communication system is used intermittently throughout this phase. It is estimated that the telemetry system will be in operation three out of each 24 hours and the voice communications five out of each 24 hours.

4 The guidance and navigation system is shut down.

5. The electrical power system is operating during this phase

6. The stabilization and control system is shut down (Manual reaction control system operation for maintenance of a slow change in the sun orientation of the vehicle is monitored visually)

7. The oxygen cells of the electrical power system are purged three times daily (every 8 hours) and the hydrogen cells twice daily (every 12 hours).

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Table 5. System Operations, Orbital Operations Phase, First Day Through Twenty-Second Day

System	Time						
	Start - Stop	Duration			Cumulative		
		Hr	Min	Sec	Hr	Min	Sec
Environmental control		528			534	20	44
Communications							
Standby	Voice communications	418 hours					
	Telemetry system	462 hours					
In use	Voice communications	110 hours			534	20	44
	Telemetry system	66 hours					
Guidance and navigation	- -				5	55	1'0
Electrical power	- -	528			536	20	44
Fuel cell purge							
Oxygen	1 5 minutes each time, 3 times a day	1	39		1	39	
Hydrogen	1.5 minutes each time, 2 times a day	1	6		1	6	
Mission cumulative time		532 hours 20 minutes 44 seconds					

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Orbital Operations Phase Event Sequence, Twenty-Third Day 24 Hours Duration

The event sequence for orbital operations on the twenty-third day is given below. Table 6 gives system operation times.

1. On the twenty-third day, the service module propulsion unit provides ΔV thrust to restore the original orbital parameters.

2. The environmental control system is in operation during this phase.

3. The communication system is used for voice communication and for telemetry of data. It is estimated that the telemetry system will be in operation for three out of the 24 hours and the voice communications for five out of the 24 hours.

4. The guidance and navigation system is started one-half hour before the orbit correction and remains on for a complete orbit after the maneuver.

5. The electrical power system is operating during this phase.

6. The stabilization and control system is used for 5 minutes for attitude alignment during the orbit correction maneuver.

7. The oxygen cells of the electrical power system are purged three times daily (every 8 hours) and the hydrogen cells twice daily (every 12 hours).

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Table 6. System Operations, Orbital Operations Phase
Twenty-Third Day

System	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		24			558	20	44
Communication												
Standby	Voice communications						19 hours					
	Telemetry system						21 hours					
										558	20	44
In use	Voice communications						5 hours					
	Telemetry system						3 hours					
Guidance and navigation	539	30		541	30		2			7	55	10
Electrical power		-			-		24			560	20	44
Stabilization and control alignment for injection	540			540	05			5		5	9	50
Fuel cell purge												
Oxygen	3 times a day							4	30	1	43	30
Hydrogen	2 times a day							3		1	9	
Propulsion - service module	540	5		540	05	05			05			25
Mission cumulative time -										556 hours 20 minutes 44 seconds		

Orbital Operations Event Sequence - 24th Day Through 45th Day (528 Hours Duration

The event sequence for the orbital operations is given below. Table 7 gives system operation times.

1. The 24th through 45th days of the mission are spent routinely in completing the remaining mission objectives.

2. The environmental control system is in operation during the whole phase.

3. The communication system is used for voice communication and for telemetry of data. It is estimated that the telemetry system will be in operation three out of each 24 hours and the voice communications five out of each 24 hours.

4. The guidance and navigation system is shut down.

5. The electrical power system is operating during this phase.

6. The O_2 cells of the electrical power system are purged on the average three times (every 8 hours) and the H_2 cells twice (every 12 hours) daily.

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Table 7. System Operations, Orbital Operations Phase, 24th Day Through 45th Day

System	Time							
			Duration			Cumulative		
	Start	Stop	Hr	Min	Sec	Hr	Min	Sec
Environmental control	-	-	528			1086	20	44
Communication								
(standby)	Voice communications		418					
	Telemetry system		462			1086	20	44
(In use)	Voice communications		110					
	Telemetry system		66					
Guidance and navigation	-	-				7	55	10
Electrical power			528			1088	20	44
Fuel cell purge O_2	Three times per day		1	39		3	27	30
H_2	Two times per day		1	6		2	15	
Mission cumulative time			1084 hours 20 minutes 44 seconds					

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Preparation for Deorbit Event Sequence (1 Hour 54 Minutes Duration)

The event sequence for the preparation for deorbit is given below. Table 8 gives the system operation times.

1. Upon completion of the mission objectives, the crew removes from the laboratory all items of equipment and data that are to be returned to earth and stores them in the command module. GOSS will have been notified of the impending reentry maneuver, and timing computations will have been made for retrograde firing of the service module engine. The on-board systems are checked out, laboratory module umbilicals and latches are disconnected, and the lab module is jettisoned. The command and service module is then prepared for entry by activating the command module electrical power and environmental control systems and by positioning the spacecraft into the proper attitude for retrofire.

2. The environmental control system is in operation during this phase.

3. The communication system is used intermittently for voice communication.

4. The guidance and navigation system is started 30 minutes before scheduled retrofire and determines command and service module retrograde attitude and positioning requirements.

5. The electrical power system is operating during this phase.

6. The S&C system is used during the separation of the command and service module from the laboratory module, following jettison of the laboratory, and also during the positioning of the command and service module for retrofiring. The system is on standby between these events.

7. The fuel cells are purged just prior to reentry.

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Table 8. System Operations, Preparation for Deorbit

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		1	54		1088	14	44
Communication												
(standby)		-			-			54		1088	14	44
(intermittent use - voice)		-			-		1					
Guidance and navigation	1085	44	44	1086	14	44		30		8	25	10
Electrical power		-			-		1	54		1090	14	44
S&C laboratory module separation	1085	50		1085	53			3				
Standby	1085	53		1086	9	44		6	44	5	24	34
Attitude alignment before retrofire	1086	9	44	1086	14	44		5				
Fuel cell purge O_2	1085			1085	1	30		1	30	3	24	
H_2	1085			1085	1	30		1	30	2	16	30
Mission cumulative time	1086 hours 14 minutes 44 seconds											

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Deorbit Operations Event Sequence (15 Minutes Duration)

The event sequence for deorbit operations is given below. Table 9 gives the system operation times

1. The crew receives the necessary data for retrofire timing and is prepared for entry GOSS is notified of readiness for retrofire. The service module engine is ignited, and the retrofiring is accomplished. Upon verification of reentry parameters by GOSS, the service module is jettisoned. The service module reaction control system is programmed to cause a translational separation of the two modules after service module jettison. The main environmental control system is in operation during this phase.

2. The CM environmental control system also is in operation.

3. The communication system is used for voice communication during this phase on an intermittent basis. GOSS is notified of readiness for and completion of retrofire and of SM jettison GOSS verifies entry parameters.

4 The guidance and navigation system is operating throughout this phase to ascertain certain parameters which are compared with those calculated by GOSS.

5. The main electrical power system is operating during this phase.

6. The CM electrical power system (batteries) are in operation.

7. The S&C system is on standby until used during the positioning of the CSM prior to service module jettison and during the separation of the two modules

8. The service module propulsion system provides the retrograde thrust for reentry.

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Table 9. System Operations, Deorbit Operations

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Main environmental control		-			-			15		1088	29	44
CM environmental		-			-			15			15	
Communication												
(Standby)		-			-			5		1088	29	44
(In use)		-			-			10				
Guidance and navigation		-			-			15		8	40	10
Main electrical power		-			-			15		1090	29	44
CM electrical power		-			-			15			15	
S&C: Standby during descent	1086	14	44	1086	23	44		9				
Attitude alignment	1086	23	44	1086	28	44		5		5	39	34
Standby during SM jettison	1086	28	44	1086	29	44		1				
Propulsion SM	1086	15		1086	15	18			18			43
Mission cumulative time	1086 hours 29 minutes 44 seconds											

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Entry Operations Event Sequence (11 Minutes Duration)

The event sequence for entry operations is given below. Table 10 gives the system operation times.

1. After jettison of the service module, the command module is maintained in a previously calculated glide path Following entry and aerodynamic slowdown, GOSS is informed of the jettisoning of the forward heat shield

2. The CM environmental control system is in operation during this phase.

3 The communication system is used for voice communications during this phase on an intermittent basis.

4. The guidance and navigation system is used during this phase to automatically control roll, pitch, and yaw to insure attainment of the programmed landing site

5 The batteries located in the CM are operating during this phase.

6. The S&C system is used intermittently as the command module is automatically maneuvered to remain within the programmed limits of the entry parameters. The system is on standby for the rest of the time in this phase

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Table 10. System Operations, Entry Operations

System	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
CM environmental control		-			-			11			26	
Communication												
(Standby)		-			-			8	30	1088	40	44
(In use)								2	30			
Guidance and navigation		-			-			11		8	51	10
CM electrical power		-			-			11			26	
S&C	(Standby and intermittent operations)							11		5	50	34
Forward heat shield jettison	1086	40	42	1086	40	44			2			2
Mission cumulative time	1086 hours 40 minutes 44 seconds											

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Parachute Descent Event Sequence (10 Minutes 16 Seconds duration)

The event sequence for parachute descent is given below. Table 11 gives the system operation times, and Table 12 gives a summary of the system operating time for the entire mission.

1. The drogue parachute is deployed followed by the main parachute release and deployment. Upon ground contact, the main parachute is released from the command module.

2. The command module enviromental control system is in operation during this phase.

3. The communication system voice transmission is used to notify GOSS of deployment of the drogue and main parachutes.

4. The guidance and navigation system is not operating during this phase.

5. The command module electrical power system is operating during this phase.

6. The S&C system is not operating during this phase.

7. The drogue parachute is deployed over a 30-second period.

8. The main parachute is deployed and activated over a period of 9 minutes and 40 seconds.

9. Following touchdown, the main parachute is released, and the recovery communication systems are activated.

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Table 11. System Operations, Parachute Descent

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
CM environmental control								10	16		36	16
Communication (Standby)								08	16			
										1088	51	
(In use)								02				
CM electrical power								10	16		36	16
Drogue parachute	1086	40	49	1086	41	19			30			30
Main parachute	1086	41	19	1086 (touchdown)	50	59			40			40
Main parachute release	1086	50	59	1086	51				1			1
Mission cumulative time										1086 hours 51 minutes		

[redacted]

Table 12. Summary of System Operating Times for Entire Mission (1086 Hours 51 Minutes Duration)

Systems	Cumulative Operating Time		
	Hours	Minutes	Seconds
Main environmental control	1088	29	44
CM environmental control		36	16
Communications	1088	51	
Guidance and navigation	8	51	10
Main electrical power	1090	29	44
CM electrical power		36	16
Stabilization and control	5	50	34
S-I propulsion		2	33
S-IVB propulsion		7	47
SM propulsion			43
Launch vehicle emergency detection		3	3
Launch escape		2	49
Fuel cell purge O_2	3	24	
H_2	2	16	30
Drogue parachute			30
Main parachute			40

APOLLO X EARTH MISSION, CSM, LOW-INCLINATION ORBIT

All Phases Event Sequence (1086 Hours 51 Minutes Duration)

The event sequence for all system operations phases is given below. Table 13 gives the system operation times for the entire mission.

The event sequence is exactly the same as for the low inclination earth mission with a laboratory module, with the following exceptions:

1. The CSM transposition and docking operations in the 80-nautical-mile parking orbit are deleted. The S&C operating time is not reduced as a result of this deletion.

2. The laboratory module separation sequence is deleted from the deorbit preparation phase. As a result of this deletion, the S&C operating time is reduced by 3 minutes.

3. Events involving the G&N system are deleted as this system is removed, and the navigation is accomplished through data link to GOSS.

Table 13 Summary of System Operating Times for Entire Mission

Systems	Cumulative Operating Time		
	Hours	Minutes	Seconds
Main environmental control	1088	29	44
CM environmental control		36	16
Communications	1088	51	
Main electrical power	1090	29	44
CM electrical power		36	16
Stabilization and control	5	47	34
S-I propulsion		2	33
S-IVB propulsion		7	47
SM propulsion			43
Launch vehicle emergency detection		3	3
Launch escape		2	49
Fuel cell purge: O_2	3	24	
H_2	2	16	30
Drogue parachute		0	30
Main parachute		9	40

APOLLO X EARTH MISSION, CSM WITH LABORATORY MODULE, POLAR ORBIT

Launch Boost Phase Event Sequence (10 Minutes 40 Seconds Duration)

The event sequence for the launch boost phase is given below. Table 14 gives the system operation times.

1. The S-IC propulsion system burns for 2 minutes and 30 seconds and is separated. The S-II ignites 1 second later and burns for 6 minutes and 32 seconds and is separated. The S-IVB ignites 1 second later and burns for 1 minute and 36 seconds to inject the spacecraft into a 100-nautical-mile parking orbit.

2. The environmental control system is put into operation 2 hours before launch and remains in operation thereafter.

3. The communications system is put on standby 2 hours before launch following prelaunch checkout.

4. The guidance and navigation system is put on standby 2 hours before launch following prelaunch checkout.

5. The S&C system is put on standby 2 hours before launch following prelaunch checkout.

6. The electrical power system is activated 4 hours before launch and remains in operation thereafter.

7. The launch vehicle emergency detection system is activated 30 seconds before launch and is used only during the first stage operations.

8. The launch escape system is operating during the initial portion of the boost phase. The tower is jettisoned 2 minutes and 54 seconds following launch.

Table 14. System Operations, Launch Boost Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hrs	Min	Sec	Hrs	Min	Sec	Hrs	Min	Sec	Hrs	Min	Sec
Environmental control	2						2	10	40	2	10	40
Communication	2						2	10	40	2	10	40
Guidance and navigation	2						2	10	40	2	10	40
S&C	2						2	10	40	2	10	40
Electrical power	4						4	10	40	4	10	40
Launch vehicle emergency detection			30		2	30		3			3	
Launch escape					2	54		2	54		2	54
Propulsion S-IC					2	30		2	30		2	30
S-II		2	31		9	3		6	32		6	32
S-IVB (First burn)		9	4		10	40		1	36		1	36
Mission cumulative time										10 minutes 40 seconds		

Orbit Attainment Phase Event Sequence (57 Minutes 40 Seconds Duration)

The event sequence for the orbit attainment phase is given below. Table 15 gives the system operation times.

1. Following S-IVB shutdown there is a coast period in parking orbit of 9 minutes and 20 seconds, during which a safety system check is conducted. The vehicle is then aligned in the correct attitude for transfer to the final orbit and the S-IVB is reignited for 6 minutes and 32 seconds followed by a coast of 39 minutes and 28 seconds, during which the command space module separates from the laboratory module, transposes, and docks. The S-IVB is then separated. At the end of the coast period, the service module engine is fired for 2 minutes and 20 seconds, resulting in a circular polar orbit at 200-nautical-mile altitude.

2. The environmental control system is in opreation during this phase.

3. Communications are used frequently throughout the phase.

4. The guidance and navigation system is used to obtain required vehicle attitude prior to S-IVB and space module engine firings and is on standby during the remainder of the phase.

5. The electrical power system is operating during this phase.

6. The S&C is used during attitude alignment prior to the S-IVB and SM engine firings and is on standby during the remainder of the phase.

7. The propulsion system of the service module is used to inject the vehicle into circular polar orbit at 200-nautical-mile altitude.

Table 15. System Operations, Orbit Attainment Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control								57	40	3	8	20
Communications								57	40	3	8	20
Guidance and navigation								57	40	3	8	20
S&C: standby								32	40			
Attitude alignment prior to S-IVB firing		15			20			5		3	8	20
Transposition, docking, and S-IV separation		27			42			15				
Attitude alignment prior to Space Module engine firing	1	1		1	6			5				
Electrical power								57	40	5	8	20
Skirt structure jettison		27			27	1		0	1			
Propulsion S-IVB (second burn)		20			26	32		6	32		8	8
Space Module	1	6		1	8	20		2	20		2	20
Mission cumulative time	one hour 8 minutes 20 seconds											

Remaining Mission Phases Event Sequence (1084 Hours 25 Minutes 50 Seconds Duration

The remaining events of the mission are exactly the same as for the low inclination earth orbit mission with laboratory module. Table 16 gives the system operation times for the remaining mission phases; Table 17 summarizes the operation times for the entire mission.

Table 16. System Operations Remaining Mission Phases

Systems	Time					
	Duration			Cumulative		
	Hrs	Min	Sec	Hrs	Min	Sec
Main environmental control	1084	4	34	1087	12	54
CM environmental control		36	16		36	16
Communication	1084	25	50	1087	34	10
Guidance and navigation	4	26		7	34	20
Main electrical power	1084	4	34	1089	12	54
CM electrical power		36	16		36	16
S&C	1	25	34	4	33	54
S-IC propulsion					2	30
S-II propulsion					6	32
S-IVB propulsion					8	08
SM propulsion			23		2	43
Launch vehicle emergency detection					3	
Launch escape					2	54
Fuel cell purge O_2	3	24		3	24	
H_2	2	16		2	16	
Drogue parachute			30			30
Main parachute		9	40		9	40

Table 17. Summary of System Operating Times for Entire Mission (1085 Hours 23 Minutes 30 Seconds Duration)

Systems	Cumulative Operating Time		
	Hours	Minutes	Seconds
Main environmental control	1087	12	54
Command module environmental control		36	16
Communication	1087	34	10
Guidance and navigation	7	34	20
Main electrical power	1089	12	54
Command module electrical power		36	16
Stabilization and control	4	35	54
S-IC propulsion		2	30
S-II propulsion		6	32
S-IVB Propulsion		8	08
Space module propulsion		2	43
Launch vehicle emergency detection		3	
Launch escape		2	54
Fuel cell purge O_2	3	24	
H_2	2	16	
Drogue parachute			30
Main Parachute		9	40

APOLLO X EARTH MISSION, CSM, POLAR ORBIT

Launch Boost Phase Event Sequence (10 Minutes 8 Seconds Duration)

The event sequence of the launch boost phase is given below. Table 18 gives the system operation times.

1. The S-IC propulsion system, burns for 2 minutes and 30 seconds and is separated. The S-II ignites 1 second later and burns for 6 minutes and 32 seconds and is separated. The S-IVB ignites 1 second later and burns for 1 minute and 4 seconds to inject the spacecraft into a 100-nautical-mile parking orbit.

2. The environmental control system is put into operation 2 hours before launch and remains in operation thereafter.

3. The communications system is put on standby 2 hours before launch following prelaunch checkout.

4. The guidance and navigation system is put on standby 2 hours before launch following prelaunch checkout.

5. The S&C system is put on standby 2 hours before launch following prelaunch checkout.

6. The electrical power system is activated 4 hours before launch and remains in operation thereafter.

7. The launch vehicle emergency detection system is activated 30 seconds before launch and is used only during the first stage operations.

8. The launch escape system is operating during the initial portion of the boost phase. The tower is jettisoned 2 minutes and 54 seconds following launch.

Table 18. Systems Operations, Launch Boost Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control	2						2	10	8	2	10	8
Communication	2						2	10	8	2	10	8
Guidance and navigation	2						2	10	8	2	10	8
S&C	2						2	10	8	2	10	8
Electrical power	4						4	10	8	4	10	8
Launch vehicle emergency detection			30		2	30		3			3	
Launch escape					2	54		2	54		2	54
Propulsion S-IC					2	30		2	30		2	30
S-II		2	31		9	3		6	32		6	32
S-IVB (1st burn)		9	4		10	8		1	4		1	36
Mission cumulative time										10 minutes 8 seconds		

Orbit Attainment Phase Event Sequence. (59 Minutes 24 Seconds Duration)

The event sequence for the orbit attainment phase is given below. Table 19 gives system operation times.

1. Following S-IVB shutdown there is a coast period in parking orbit of 9 minutes and 52 seconds during which a safety system check is conducted. The vehicle is then aligned in the correct attitude for transfer to the final orbit, and the S-IVB is ignited for the second time, burning for 3 minutes and 32 seconds. The vehicle coasts for 39 minutes and 28 seconds before the S-IVB is ignited for the third time with a burn time of 3 minutes and 32 seconds. This results in a circular polar orbit at 200-nautical-mile altitude. The skirt structure is then jettisoned, and the S-IVB is separated.

2. The environmental control system is in operation during this phase.

3. Communications are used frequently throughout the phase.

4. The guidance and navigation system is used to obtain required vehicle attitudes prior to S-IVB firings and is on standby during the remainder of the phase.

5. The S&C system is used for attitude alignment prior to S-IVB firings and during the S-IVB separation. The system is on standby during the remainder of the phase.

6. The electrical power system is operating during this phase.

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Table 19. Systems Operations, Orbit Attainment Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control								59	24	3	9	32
Communication								59	24	3	9	32
Guidance and navigation								59	24	3	9	32
S&C: Attitude alignment prior to second S-IVB burn		15			20			5				
Attitude alignment prior to third S-IVB burn		58		1	3			5		3	9	32
S-IVB separation	1	6	32	1	9	32		3				
Standby								46	24			
Electrical power								56	24	5	9	32
Skirt structure jettison	1	6	32	1	6	33			1			
Propulsion S-IVB (second burn)		20			23	32		3	32			
S-IVB (third burn)	1	3		1	6	32		3	32		8	40
Mission cumulative time								1 hour 9 minutes 32 seconds				

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Remaining Mission Phases (1084 (1084 Hours 25 Minutes 50 Seconds Duration)

The remaining events of the mission are exactly the same as for the polar orbit earth mission with a laboratory module with the following exception: the laboratory module separation operation in the deorbit preparation phase is deleted, and, as a result, the S&C system operating time is reduced by 3 minutes. Table 20 gives the system operation times for the remaining mission phases, and Table 21 gives the operation times for the entire mission.

Table 20. System Operations Remaining Mission Phases

Systems	Time					
	Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec
Main environmental control	1084	4	34	1087	14	6
Command module environmental control		36	16		36	16
Communication	1084	25	50	1087	35	22
Guidance and navigation	4	26		7	35	32
Main electrical power	1084	4	34	1089	14	6
Command module electrical power		36	16		36	16
S&C	1	22	34	4	32	6
S-IC propulsion					2	30
S-II propulsion					6	32
S-IVB propulsion					8	4
Space module propulsion			23			23
Launch vehicle emergency detection					3	
Launch escape					2	24
Fuel cell purge O_2	3	24		3	24	
H_2	2	16		2	16	
Drogue parachute			30			30
Main parachute		9	40		9	40

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Table 21. Summary of System Operation Times for Entire Mission

Systems	Cumulative Operating Times		
	Hours	Minutes	Seconds
Main environmental control	1087	14	6
Command module environmental control		36	16
Communication	1087	35	22
Guidance and navigation	7	35	32
Main electrical power	1089	14	6
Command module electrical power		36	16
Stabilization and control	4	32	6
S-IC propulsion		2	30
S-II propulsion		6	32
S-IVB propulsion		8	4
Space module propulsion			23
Launch vehicle emergency detection		3	
Launch escape		2	24
Fuel cell purge O_2	3	24	
H_2	2	16	
Drogue parachute			30
Main parachute		9	40

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APOLLO X 28-DAY LUNAR MAPPING MISSION, CSM WITH LABORATORY MODULE, POLAR LUNAR ORBIT

Ascent Phase, Event Sequence (11 Minutes 46 Seconds Duration)

The event sequence for the ascent phase is given below. Table 22 gives the system operation times.

1. The S-IC propulsion burns for 2 minutes and 28 seconds and is separated. The S-II ignites 2 seconds later and burns for 6 minutes and 34 seconds. After a 3-second delay for staging, the S-IVB is ignited and burns for 2 minutes and 39 seconds to inject the spacecraft into a 100 nautical mile parking orbit.

2. The environmental control system is put into operation 2 hours before launch and remains in operation thereafter.

3. The communication system is put on standby 2 hours before launch (following prelaunch checkout) and remains in operation thereafter.

4. The guidance and navigation system is put on standby 2 hours before launch (following prelaunch checkout) and remains in operation thereafter.

5. The S&C system is put on standby 2 hours before launch (following prelaunch checkout) and remains in operation thereafter.

6. The electrical power system is activated 4 hours before launch and remains in operation thereafter.

7. The launch vehicle emergency detection system is activated 30 seconds before launch and is used only during the first stage operations.

8. The launch escape system is operating during the initial portion of the boost phase. The tower is jettisoned 3 minutes and 10 seconds following launch.

Table 22. System Operations, Ascent Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control	-2	00	00			-	2	11	46	2	11	46
Communication	-2	00	00			-	2	11	46	2	11	46
Guidance and navigation	-2	00	00			-	2	11	46	2	11	46
S & C	-2	00	00			-	2	11	46	2	11	46
Electrical power	-4	00	00			-	4	11	46	4	11	46
Launch vehicle emergency detection			-30		2	28		2	58		2	58
Launch escape					3	10		3	10		3	10
Propulsion:												
S-IC					2	28		2	28		2	28
S-II		2	30		9	4		6	34		6	34
S-IV (first burn)		9	7		11	46		2	39		2	39
Mission cumulative time	11 minutes 46 seconds											

Earth-Orbital Phase, Event Sequence (2 Hours 24 Minutes 43 Seconds Duration)

The event sequence for the earth-orbital phase is given below. Table 23 gives the skstem operation times.

1. Following parking orbit attainment, the computation of orbit parameters and comparison with GOSS data requires 2 minutes. A spacecraft systems check is then conducted requiring 9 minutes and 36 seconds. Subsequent navigation operations consume 1 hour 43 minutes 24 seconds. The remaining 31 minutes and 43 seconds of the phase are used for a complete systems check.

2. The environmental control system is in operation during this phase.

3. The communication system is used frequently throughout this phase.

4. The guidance and navigation system is in use throughout the navigation sequence and is on standby for the remainder of the phase.

5. The S & C system is used for attitude alignments during the navigation sequence and is on standby for the remainder of the phase.

6. The electrical power system is in operation during the entire phase.

Table 23 System Operations, Earth-Orbital Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control			-			-	2	24	43	4	36	29
Communication			-			-	2	24	43	4	36	29
Guidance and navigation:												
Standby		11	46		21	22		9	36			
Navigation		21	22	2	4	46	1	43	24	4	36	29
Standby	2	4	46	2	36	29		31	43			
S & C:												
Standby			-			-	2	19	43			
Attitude alignments			-			-		5		4	36	29
Electrical power			-			-	2	24	43	6	36	29
Mission cumulative time	2 hours 36 minutes 29 seconds											

Translunar Injection Phase, Event Sequence (28 Minutes 30 Seconds Duration)

The event sequence for the translunar injection phase is given below. Table 24 gives the system operation times.

1. Translunar injection parameters are computed and compared with GOSS data. The IMU is fine-aligned and checked. The stabilization and control, guidance and navigation, and service module reaction control systems are readied for the lunar injection, and the vehicle is properly oriented. The S-IVB engine is then fired by a signal from the guidance computer to inject the spacecraft into translunar orbit.

2. The environmental control system is in operation throughout the phase.

3. The communications system is used for injection parameter verification. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is operating throughout the phase for IMU fine alignment, fine-alignment check, and for the injection maneuvers.

5. The electrical power system is operating during this phase.

6. The stabilization and control system is used for vehicle attitude alignment during the IMU alignment and check and during the injection maneuvers. The system is on standby during the remainder of the phase.

7. The propulsion system of the S-IVB is used for 5 minutes and 10 seconds to inject the spacecraft into translunar orbit.

Table 24. System Operations, Translunar Injection Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control								28	30	5	4	59
Communications								28	30	5	4	59
Guidance and navigation								28	30	5	4	59
Electrical power								28	30	7	4	59
Stabilization and control								28	30	5	4	59
Propulsion:												
S-IVB (second burn)	2	59	49	3	4	59		5	10		7	49
Mission cumulative time	3 hours 4 minutes 59 seconds											

Initial Translunar Coast and Transposition Phase, Event Sequence (1 Hour Duration)

The event sequence for the initial translunar coast and transposition phase is given below. Table 25 gives the system operation times.

1. After injection there is a 15-minute coast period during which trajectory parameters are verified with GOSS and all systems are checked. The next 15 minutes are used to prepare for and implement the CSM-LAB/S-IVB transposition. After the preparations, including severence of the LEM adapter, have been completed, the CSM is separated from the LAB/S-IVB, rotated 180 degrees in pitch and docked with the LAB/S-IVB. The CSM/LAB and S-IVB are then separated by the service module reaction control system.

2. The environmental control system is in operation during this phase.

3. The communication system is used to verify trajectory parameters and to confirm readiness for transposition. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is on standby during this phase.

5. The electrical power system is operating during this phase.

6. The S&C system is used for attitude hold during LEM adapter severance and during the transposition/S-IVB separation sequence. The system is on standby during the remainder of the phase.

Table 25. System Operations, Initial Translunar Coast and Transposition Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control							1			6	4	59
Communication							1			6	4	59
Guidance and navigation - standby							1			6	4	59
Electrical power							1			6	4	59
Stabilization and control:												
Standby	3	4	59	3	19	59		15				
LEM adapter severance	3	19	59	3	24	59		5				
Standby	3	24	59	3	34	59		10		6	4	59
Transposition/ S-IVB separation	3	34	59	4	4	59		30				
Mission cumulative time	4 hours 4 minutes 59 seconds											

Translunar Phase, Event Sequence (70 Hours 22 Minutes 3 Seconds Duration)

The event sequence for the translunar phase is given below. Table 26 gives the system operation times.

1. Following S-IVB separation the high-gain antennas are deployed, and the crew couches are repositioned. The laboratory module is pressurized, laboratory systems are activated, umbilicals are connected, and all spacecraft systems are checked. The normal work-rest cycle is then initiated. The first, second, and third midcourse corrections are made 7 hours, 50 hours 50 minutes, and 70 hours 14 minutes, respectively, after the start of the phase. The midcourse corrections are accomplished by the gimballed service module main engine after required velocity corrections in magnitude and direction have been determined from GOSS navigational data. System checkouts and fuel cell purges are performed periodically as time is available throughout the phase.

2. The environmental control system is in operation during this phase.

3. The communication system is used intermittently for voice and telemetry of data. In particular, navigational and trajectory data is obtained from GOSS before and after each course correction. The system is on standby during the remainder of the phase.

4. The guidance and navigational system is shut down after the first system check (30 minutes after the start of the phase). It is reactivated for a 30-minute interval centered at each service module engine firing for IMU fine alignment, alignment check, insertion of navigational data, service module engine fire control, and for delta V verification.

5. The electrical power system is operating during this phase.

6. The S&C system is shut down after the first system check. It is reactivated for a 30-minute interval centered at each service module engine firing for attitude alignment, ullage acceleration, and attitude hold during service module engine firing.

7. The service module propulsion system is used for three separate midcourse corrections of seven seconds each.

8. The O_2 cells of the electrical power system are purged on the average of three times (every 8 hours) and the H_2 cells twice (every 12 hours) daily.

Table 26. System Operations, Translunar Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control							70	22	3	76	27	2
Communication							70	22	3	76	27	2
Guidance and navigation:												
Standby	4	4	59	4	34	59		30				
First midcourse correction	10	50		11	20			30				
Second midcourse correction	54	40		55	10			30		7	58	1
Third midcourse correction	74	4		74	27	2		23	2			
Electrical power							70	22	3	78	28	2

Table 26. System Operations, Translunar Phase (Cont)

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Stabilization and control:												
Standby	4	4	59	4	34	59		30				
First midcourse correction	10	50		11	20			30				
Second midcourse correction	54	40		55	10			30		7	58	1
Third midcourse correction	74	4		74	27	2		23	2			
Propulsion:												
service module first burn	11	4	59	11	5	6			7			
service module second burn	54	54	59	54	55	6			7			21
service module third burn	74	18	59	74	19	6			7			
Fuel cell purge												
O_2	1.5 minutes - three times a day										13	30
H_2	1.5 minutes - two times a day										9	
Mission cumulative time	74 hours 27 minutes 2 seconds											

Lunar-Orbital Injection Phase Sequence of Events (1 Hour 21 Minutes 49 Seconds Duration)

The event sequence for the lunar-orbital injection phase is given below. Table 27 gives the system operation times.

1. Upon completion of the final midcourse correction maneuver all spacecraft systems are checked and preparations are completed for injection into an 80-nautical mile lunar polar orbit. The injection is accomplished by the gimballed service module main engine after the required delta velocity vector has been computed from GOSS navigational data.

2. The environmental control system is in operation during this phase.

3. The communication system is used to obtain GOSS navigational data before injection. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is used during the first part of the phase for completing the verification of the third midcourse correction. Following a standby period of approximately 49 minutes, the system is used during the remainder of the phase for IMU fine alignment, insertion of navigational and injection parameters, and for service module engine fire control.

5. The electrical power system is operating during this phase.

6. The S&C system is on standby for the first 56 minutes 27 seconds of the phase. The system is used during the remainder of the phase for attitude alignment, ullage acceleration, and attitude hold during service module engine firing.

7. The service module propulsion system is used for 5 minutes and 20 seconds to inject the spacecraft into lunar orbit.

Table 27. System Operations, Lunar- Orbit Injection Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Mın	Sec
Environmental control							1	21	49	77	48	51
Communication							1	21	49	77	48	51
Guidance and navigation:												
Midcourse correction verification	74	27	2	74	34	2		7				
Standby	74	34	2	75	23	29		49	27	9	19	50
Lunar injection	75	23	29	75	48	51		25	22			
Electrical power							1	21	22	79	48	51
Stabilization and control:												
Standby	74	27	2	75	23	29		56	27			
Lunar injection	75	23	29	75	48	51		25	22	9	19	50
Propulsion:												
Service module (fourth burn)	75	43	29	75	48	49		5	20		5	41
Mission cumulative time	75 hours 48 minutes 51 seconds											

Lunar-Orbital Phase, Event Sequence (672 Hours Nominal Duration)

The event phase for the lunar-orbital phase is given below. Table 28 gives the system operation times.

1. Immediately after lunar injection, the orbit parameters are verified by GOSS and orbital corrections are made if necessary. A systems check is then conducted and preparations are made to begin the scheduled mapping operations, which require a total lunar orbit stay time of approximately 28 days, or 330 orbits, because the mapping is confined to the illuminated side of the moon. When the crew and vehicle are ready, mapping is begun at a pole and continues from pole to pole on the 6th, 12th, 18th, etc. subsequent orbits, and from 45 degrees north to 45 degrees south selenographic latitude on the 3rd, 9th, 15th, etc. subsequent orbits. The spacecraft is held to a precise local vertical orientation during mapping.

2. The environment control system is in operation during this phase.

3. The communication system is on standby when not being used for voice communication or telemetry of data. It is estimated that such use will total 5 hours out of each 24 hours.

4. The guidance and navigation system is used for 30 minutes on the 9th and the 18th days of the phase for lunar orbit corrections by the reaction control system.

5. The electrical power system is operating during this phase.

6. The stabilization and control system is started 15 minutes before each mapping period for warm up and initial alignment and is used to hold local vertical during mapping. The system is also used for 30 minutes on the 9th and the 18th days of the phase for lunar orbit corrections by the reaction control system.

7. The O_2 cells of the electrical power system are purged on the average three times (every 8 hours) and the H_2 cells twice (every 12 hours) daily.

Table 28. System Operations, Lunar-Orbital Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control							672			749	48	51
Communication							672			749	48	51
Guidance and navigation:												
Orbit correction on 9th day of phase	275	48	51	276	18	51		30				
Orbit correction on 18th day of phase	475	48	51	476	18	51		30		10	19	50
Electrical power							672			751	48	51
Stabilization and control mapping							82	30				
110 - 15-minute warmup periods							27	30				
Orbit correction on 9th day of phase	275	48	51	276	18	51		30		120	19	50
Orbit correction on 18th day of phase	475	48	51	476	18	51		30				
Fuel cell purge												
O_2	Three times daily									2	19	30
H_2	Two times daily									1	33	
Mission cumulative time	747 hours 48 minutes 51 seconds											

Transearth Injection Phase, Sequence of Events (1 Hour 54 Minutes Duration)

The event sequence for the transearth injection phase is given below. Table 29 gives the system operation times.

1. Upon completion of the mapping operations, the crew removes from the laboratory all items of equipment, film, and data that are to be returned to earth and stores these in the command module. GOSS will have been notified of the impending transearth injection maneuver, and the time for service module firing will have been computed. The on-board systems are checked out, laboratory module umbilicals and latches are disconnected, and the laboratory module is jettisoned. The command and service module is then positioned in the correct attitude for transearth injection and the service module engine is fired.

2. The environmental control system is in operation during this phase.

3. The communication system is used frequently during the phase to obtain GOSS navigational data before injection. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is started one-half hour before scheduled retrofire for IMU fine alignment, fine alignment check, and for injection maneuvers.

5. The electrical power system is operating during this phase.

6. The stabilization and control system is used during the separation of the command and service module from the laboratory module following jettison of the lab and also during the positioning of the command service module for injection. The system is on standby for 10 minutes between these two events.

7. The propulsion system is used for 3 minutes to inject the spacecraft into transearth orbit.

Table 29. System Operations, Transearth Injection Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		1	54		751	42	51
Communications		-			-		1	54		751	42	51
Guidance and navigation	749	12	51	749	42	51		30		10	49	50
Electrical power		-			-			54		753	42	51
Stabilization and control:												
Laboratory module separation	749	21	51	749	24	51		3				
Standby	749	24	51	749	34	51		10		120	40	50
Attitude alignment for injection	749	34	51	749	42	51		8				
Propulsion: Service module fifth burn	749	39	51	749	42	51		03			08	41
Mission cumulative time	749 hours 42 minutes 51 seconds											

Transearth Phase, (71 Hours Duration) Sequence of Events

The event sequence for the transearth phase is given below. Table 30 gives the system operation times.

1. Following transearth injection, the normal work-rest cycle is begun. The first midcourse correction is made 12 hours after injection as the spacecraft nears the lunar sphere of influence. The second correction is made 6 hours after the first. Twenty-seven hours after the second correction, a third correction is made and followed by a fourth and final correction 26 hours later. The midcourse corrections are accomplished by the gimballed service module main engine after required velocity corrections in magnitude and direction have been determined from GOSS navigational data. System checkouts and fuel cell purges are performed periodically as time is available throughout the phase.

2. The environmental control system is in operation during this phase.

3. The communication control system is used intermittently for voice and telemetry of data. In particular, navigational and trajectory data is obtained from GOSS after injection and before and after each course correction. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is shut down after the first system check, 30 minutes after the start of the phase. It is reactivated for a 30-minute interval centered at each service module engine firing for IMU fine alignment, fine alignment check, insertion of navigational data, service module engine fire control, and for delta V verification.

5. The electrical power system is operating during this phase.

6. The S&C system is shut down after the first system check and is reactivated for a 30-minute interval centered at each service module engine firing for attitude alignment, ullage acceleration, and attitude hold during service module engine firing.

7. The service module propulsion system is used for four separate midcourse corrections of 180 seconds, 142 seconds, 10 seconds and 10 seconds respectively.

8. The 0_2 cells of the electrical power system are purged on the average three times (every 8 hours) and the H_2 cells twice (every 12 hours) daily.

Table 30. System Operations, Transearth Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		71			822	42	51
Communication		-			-		71			822	42	51
Guidance and navigation:												
Standby	749	42	51	750	12	51		30				
First midcourse correction	761	27	50	761	57	50		30				
Second midcourse correction	767	27	50	767	57	50		30		13	05	00
Third midcourse correction	794	27	50	794	57	50		30				
Fourth midcourse correction	820	27	41	820	42	51		15	10			
Electrical power		-			-		71			824	42	51
Stabilization and control:												
Standby	749	42	51	750	12	51		30				
First midcourse correction	761	27	50	761	57	51		30				
Second midcourse correction	767	27	50	767	57	50		30		122	56	00
Third midcourse correction	794	27	50	794	57	50		30				
Fourth midcourse correction	820	27	41	820	42	51		15	10			
Service module propulsion:												
Sixth burn	761	42	50	761	45	50		3				
Seventh burn	767	42	50	767	45	12		2	22	00	14	23
Eighth burn	794	42	50	794	43	00			10			
Ninth burn	820	42	41	820	42	51			10			
Fuel Cell Purge O_2	1.5 minutes, three times a day									02	33	00
H_2	1.5 minutes, two times a day									01	42	00
Mission cumulative time	820 hours 42 minutes 51 seconds											

Preentry Phase, Event Sequence (1 Hour Duration)

The event sequences for the preentry phase is given below. Table 31 gives the system operation times.

1. Immediately after the final course correction, a complete preentry check is conducted. Entry parameters are verified by GOSS and service module separation data is exchanged and confirmed. The couches are positioned and the crewmen don pressure suits and prepare for entry. The spacecraft is positioned to separation attitude orientation, and the service module is jettisoned. The service module reaction control system is programmed to cause a translational separation of the two modules.

2. The main environmental control system is in operation during this phase.

3. The command module environmental control system is also in operation.

4. The communication system is used for voice communication during this phase on an intermittent basis. The system is on standby during the remainder of the phase.

5. The guidance and navigation system is operating throughout the phase to ascertain entry and separation parameters, which are compared with those calculated by GOSS.

6. The main electrical power system is operating during this phase.

7. The command module electrical power system (batteries) is also in operation.

8. The stabilization and control system is on standby until used during the positioning of the spacecraft prior to service module jettison and during separation of the two modules.

Table 31. System Operations Preentry Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Environmental control		-			-		1			823	42	51
Command module environmental control							1			1		
Communication							1			823	42	51
Guidance and navigation							1			14	5	
Main electrical power							1			825	42	51
Command module electrical power							1			1		
Stabilization and control:												
Standby	820	42	51	821	36	51		54				
Attitude alignment prior to service module jettison	821	36	51	820	41	51		5		123	56	
Attitude alignment during service module jettison	821	41	51	820	42	51		1				
Mission cumulative time	821 hours 42 minutes 51 seconds											

Entry Phase, Event Sequence (11 Minutes Duration)

The event sequence for the entry phase is given below. Table 32 gives the system operation times.

1. After jettison of the service module, the command module is maintained in a previously calculated glide path. Following entry and aerodynamic slowdown, GOSS is informed of forward heat shield jettison.

2. The command module environmental control system is operating during this phase.

3. The communication system is used intermittently for voice communications.

4. The guidance and navigation system is used during this phase to automatically control roll, pitch, and yaw to ensure attainment of the programmed landing site.

5. The batteries located in the command module are operating during this phase.

6. The S&C system is used intermittently as the command module is automatically maneuvered to maintain entry parameters within their programmed limits. The system is on standby during the remainder of the phase.

Table 32. System Operations, Entry Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Command module environmental control								11		1	11	
Communications								11		823	53	51
Guidance and navigation								11		14	16	
Command module electrical power								11		1	11	
Stabilization and control	(Standby and intermittent operations)							11		124	7	
Forward heat shield jettison	821	53	49	821	53	51			2			
Mission cumulative time	821 hours 53 minutes 51 seconds											

Parachute Descent Phase, Event Sequence (10 Minutes 16 Seconds Duration)

The event sequence for the parachute descent phase is given below. Table 33 gives the system operation times. Table 34 summarizes the operating times for the 34-day lunar mapping mission.

1. The drogue parachute is deployed, followed by the main parachute release and deployment. Upon ground contact, the main parachute is released from the command module.

2. The command module environmental control system is in operation during this phase.

3. The communication system is used to notify GOSS of deployment of the drogue and main parachutes. The system is on standby during the remainder of the phase.

4. The guidance and navigation system is not operating during this phase.

5. The command module electrical power system is operating during this phase.

6. The S&C system is not operating during this phase.

7. The drogue parachute is deployed over a 30-second period.

8. The main parachute is deployed and activated over a period of 9 minutes and 40 seconds.

9. Following touchdown, the main parachute is released and the recovery communications systems are activated.

Table 33. System Operations, Parachute Descent Phase

Systems	Time											
	Start			Stop			Duration			Cumulative		
	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec	Hr	Min	Sec
Command module environmental control		-			-			10	16	1	21	16
Communication		-			-			10	16	824	4	6
Command module electrical power		-			-			10	16	1	21	16
Drogue parachute	821	53	55	821	54	25			30			30
Main parachute	821	54	25	822	4	5		9	40		9	40
Main parachute release	822	4	5	822	4	6			1			1
Mission cumulative time	822 hours 4 minutes 7 seconds											

Table 34. Summary of System Operating Times For 28-day Lunar Mapping Mission (822 Hours 4 Minutes 7 Seconds Duration)

Systems	Total Operating Time		
	Hours	Minutes	Seconds
Main environmental control	823	42	51
Command module environmental control	1	21	16
Communications	824	4	6
Guidance and navigation	14	16	
Main electrical power	825	42	51
Command module electrical power	1	21	16
Stabilization and control	124	7	
S-IC propulsion		2	28
S-II propulsion		6	34
S-IVB propulsion		7	49
Service module propulsion		14	23
Launch vehicle emergency detection		2	58
Launch escape		3	10
Fuel cell purge O_2	2	33	
H_2	1	42	
Drogue parachute			30
Main parachute		9	40

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LANDING AND RECOVERY ANALYSIS (CONTINGENCY)

LOW INCLINATION ORBIT RECOVERY

A detailed emergency landing and recovery analysis for a rendezvous compatible 260-nautical-mile 29.1-degree earth orbit was made in connection with the Extended-Mission Apollo Study (S&ID 63-1370-2) and will not be repeated here. However, the previous analysis does not apply directly to the case of a 200-nautical-mile altitude orbit, because the latter orbit is not rendezvous compatible (i.e., it does not have a 24-hour repetitive earth trace).

Thus a computer program has been used to determine the probability that the spacecraft could land within at least one of three circular contingency areas of radius R centered near Okinawa, near Pearl Harbor, and off the east coast of Florida. The orbit inclination and altitude are assumed to be 28.5 degrees and 207.1 nautical miles, respectively. The latter value is used instead of 200 nautical miles, because the earth trace repeats every 3 days and thus greatly simplifies the analysis. The value of R is taken to be 1150 nautical miles. The Apollo X a lateral maneuverable capability of about 150 nautical miles, implies the recovery ships would have to travel 1000 nautical miles. In the Extended-Mission Apollo Study, the average speed of the recovery ships was assumed to be 30 knots per hour. However here the assumed average speed is 15 knots per hour to conform with later information. To be conservative, no assistance from aircraft is assumed.

Computer results show that of the 46 orbits in any 3-day period there are the 11 orbits from which a landing in one of the specified areas cannot be made; i.e., these orbits are numbered as follows: 9, 10, 11, 24, 25, 26, 27, 40, 41, 42, and 43. This is a total of 11 orbits out of 46 or a probability of 0.239 that a landing cannot be made in a safe area, assuming that at least one complete orbit is possible after an emergency occurs. With two complete orbits, the probability is clearly 8/46 or 0.174. Similarly, with three or four hold orbits, the probabilities become 0.109 and 0.044 that landing in a safe area cannot be made. For five or more hold orbits, at least one of the sites can always be reached.

In the worst case (no hold orbits and touchdown at the extreme edge of the 1000-nautical-mile circle), the crew waiting time is 1000/15 or 67 hours. This waiting-time is reduced by approximately 1.5 hours for every hold orbit if the recovery force is notified of the intent to land as soon as the emergency occurs. It is concluded that the maximum crew waiting time

under the most unfavorable conditions is below the 72-hour survival time assigned to the Apollo LOR mission, and that there is an 89-percent probability that the spacecraft can land in a contingency area if at least three hold orbits are possible. The data is summarized below:

Number of Complete Hold Orbits	Maximum Crew Waiting Period (hrs)	Probability of Landing in One of Three Contingency Areas
1	67	0.761
2	65.5	0.826
3	63	0.891
4	61.5	0.956
5	60	1.000

POLAR ORBIT RECOVERY

Nonstandby recovery forces for emergency reentry from polar orbit are not satisfactory from a crew safety or an operational viewpoint. Although two common orbit points do exist for any polar orbit (i.e., the North and South poles), NASA documents indicate that landings at greater than 45 degrees north or south latitudes would not be acceptable because of temperature and weather considerations. Fortunately, an almost unbroken band of water, roughly 1000 nautical miles wide, extends entirely around the earth near 45 degrees south latitude Thus it will be assumed that a standby force of recovery ships will be more or less equally spaced along the 45 degrees south latitude line. The total distance around the earth at this latitude is approximately 15,300 nautical miles.

The worst possible time for an emergency to occur would be shortly after the vehicle has passed 45 degrees south latitude and at such a time that a point on earth, just halfway between any two of the recovery ships, rotates into the orbital plane just as the vehicle next approaches the recovery latitude In other words, an emergency landing point will never be more than half of the ship's separation distance away from a rescue craft Furthermore, the Apollo vehicle has a lateral maneuverable capability of about 140 nautical miles (based on an L/D ration of 0.42) that can be used to further reduce the maximum rescue distance

The period of a vehicle in a 200-nautical-mile circular orbit is about 92 minutes, which is equivalent to 15.7 orbits per 24 hours Thus, during one period, a point on the earth's surface at 45 degrees south latitude will rotate by the orbital plane a distance of 15300/15.7 or approximately 970 nautical miles. If it is possible for the vehicle to stay in orbit for one or

more complete hold orbits after a decision has been made to make an emergency reentry, the maximum rescue distance will be reduced by 970 times the number of hold orbits if the earth trace of the orbit is either halfway between the ships or nearer the adjacent recovery ship to the west at the time that the space vehicle first approaches the recovery latitude (case 1). If the orbit trace is nearer the adjacent ship to the east (case 2), the rescue distance generally will be reduced somewhat less that 970 nautical miles per hold orbit and even increased in some situations. The two cases must be considered separately.

Although it is quite possible that ship-based aircraft or helicopters may be used to expedite the recovery, it will be assumed conservatively that only ships are available, and that the closest ship proceeds to the touchdown point at an average speed of 30 nautical miles per hour. Thus for every hold orbit the maximum rescue distance will be reduced by an additional 1 5, 30, or 45 nautical miles This reduction assumes, of course, that the recovery ships are notified of the intent to reenter as soon as that decision has been made

If the maximum waiting time after touchdown (hours) is denoted by T, the number of recovery ships by n, and the number of hold orbits by X, the following equation applicable to case 1 takes all of the above mentioned factors into account for recovery at 45 degrees latitude:

$$T = \frac{(15300/2n) - 140 - (970 + 45)\,X}{30}$$

Case I values of T are shown in the applicable cells of the matrix, Table 35.

Table 35. Crew Waiting Time Matrix

	n						
X*	1	2	3	4	5	6	7
0	250. 3	122 8	80 3	59 1	46 3	37 8	31 7
1	216. 5	89. 0	46 5	25. 3	12. 5	4 0	0
2	182. 7	55 2	12 7	0	0	0	0
3	148 9	21. 4	0	0	0	0	0
4	115 1	0	0	0	0	0	0

Note: A zero value for X means that the vehicle lands at 45 degrees latitude at the first opportunity; a value of 1 for x means that the vehicle waits until the second opportunity, and so on

Based on a maximum acceptable value of 72 hours survival time after landing, it is seen that for case 1 at least two ships must be used for two, three, or four assumed hold orbits while at least three and four ships are required for an assumption of one and no hold orbits, respectively. It is quite likely that at least four standby recovery ships would be deployed, in which case it is seen that there would be no waiting time if two hold orbits can be tolerated, and at worst, only 25.3 hours if one hold orbit can be tolerated If as many as six recovery ships were available, the maximum time for one hold orbit would be reduced to only 4.0 hours.

Case II requires a different analysis. If the earth trace of the orbit as the vehicle first approaches the recovery latitude (after a decision has been made to reenter) is 970/2 or 485 nautical miles east of the midpoint between two of the recovery ships, it is evident that the touchdown point will be as far from a (fixed) recovery ship after one hold orbit as it is with no hold orbits. If the 45 nautical miles traveled by a ship during one orbit is considered, the critical earth trace intercept point is (970 + 45)/2 or 507.5 nautical miles east of the midpoint between the ships.

If the distance from the earth trace intercept point on the recovery latitude line is less than 507.5 nautical miles, one hold orbit will result in a reduction in crew waiting time. At the midpoint the reduction is equal to 33.8 hours, which is just the difference between the corresponding elements of row 1 and row 2 in the T matrix of case 1.

Similarly, for two hold orbits, the critical earth trace intercept point where there is no reduction in crew waiting time is 2 (970) + 45/2 or 1015 nautical miles east of the ship's midpoint. At the midpoint, the reduction amounts to 67.6 hours for two hold orbits, which is the difference between corresponding elements of row 1 and row 3 in the T matrix.

For three hold orbits, the earth trace intercept point corresponding to no reduction in waiting time is 1522.5 nautical miles east of the ship's midpoint. At the midpoint the reduction resulting from three hold orbits is 101.4 hours or the difference between the corresponding elements of row 1 and row 4 of the T matrix, etc.

The above relationships are shown in Figure 4 which might be used as follows. Immediately subsequent to a decision to make an emergency reentry, the estimated (first approach) earth trace intercept distance to the recovery ships' midpoint is either determined by the Apollo X crew or transmitted from an earth station. If this distance is greater than 507.5 nautical miles, and if it is decided that only one hold orbit could be tolerated, then reentry should be made at the first opportunity (no hold orbit). If the distance is less than 507.5 nautical miles, or if two or more hold orbits could be tolerated, the crew must balance the reductions in waiting time as shown

in Figure 4 against the possible risks or dangers associated with the corresponding hold orbits. Similar decision processes apply for distances of 1015.0 and 1522.5 nautical miles when two or three hold orbits, respectively, are decided to be the most that could be tolerated.

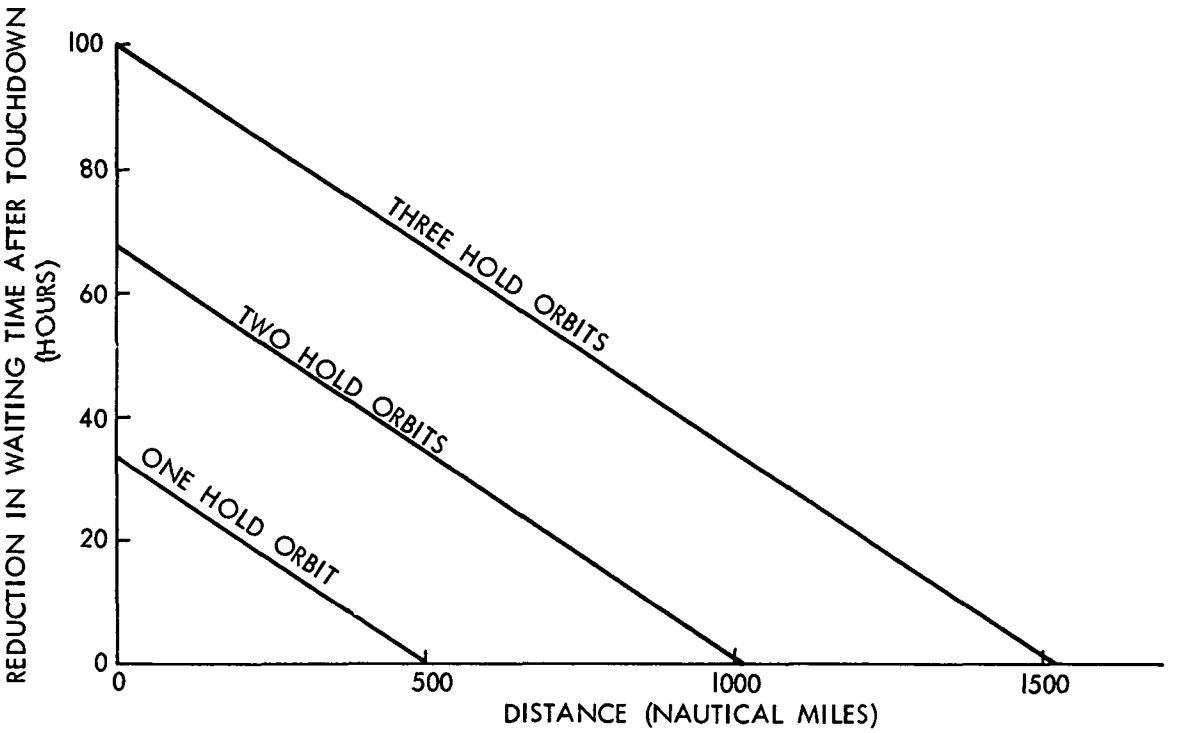

Figure 4. Reduction in Crew Waiting Time as a Function of Earth Orbit Trace Position (Case 2)

Results applicable to both case 1 and case 2 are combined in Figures 2 and 3 for four and six equally spaced recovery ships, respectively. This combination is made possible by considerating the distance from the Apollo X (first approach) earth orbital trace to the adjacent recovery ship on the west as the independent variable and then using data from Table 35 and Figure 4.

It is seen from Figure 5 that, irrespective of the position of the Apollo X earth trace relative to the recovery ships, the maximum crew waiting time after touchdown in the case of 4 recovery ships is 59.1, 44.0, or 28.0 hours for zero, one, or two hold orbits, respectively.

Similarly, from Figure 6 it is evident that, for six recovery ships, the maximum crew waiting time is 37.8 or 22.7 hours for zero or one hold orbits, respectively.

Since all of these worst-case values are well below the assumed 72-hour Apollo X survival limit, it is concluded that either 4 or 6 recovery ships at 45 degree south latitude are certainly adequate for crew recovery after unscheduled reentry from polar orbit. Furthermore, tnere would be numerous cases when emergency landings could be made safely at other points than at 45 degree south latitude. This might be desirable not only when excessive hazard prevented orbit hold time to reach the 45-degree south latitude line but also because of the inherent desirability of a special site (for example a site in tne Gulf of Mexico). Tnese considerations imply that, in general, some further reduction in expected waiting time can be assumed from the values calculated in this analysis.

In addition to the craft mentioned above, it will be desirable to have several emergency recovery ships available during the launch operation. Since the probability of mission abort is greatest at stage ignition, these ships should be located where landing would be most likely to occur in case of ignition failure.

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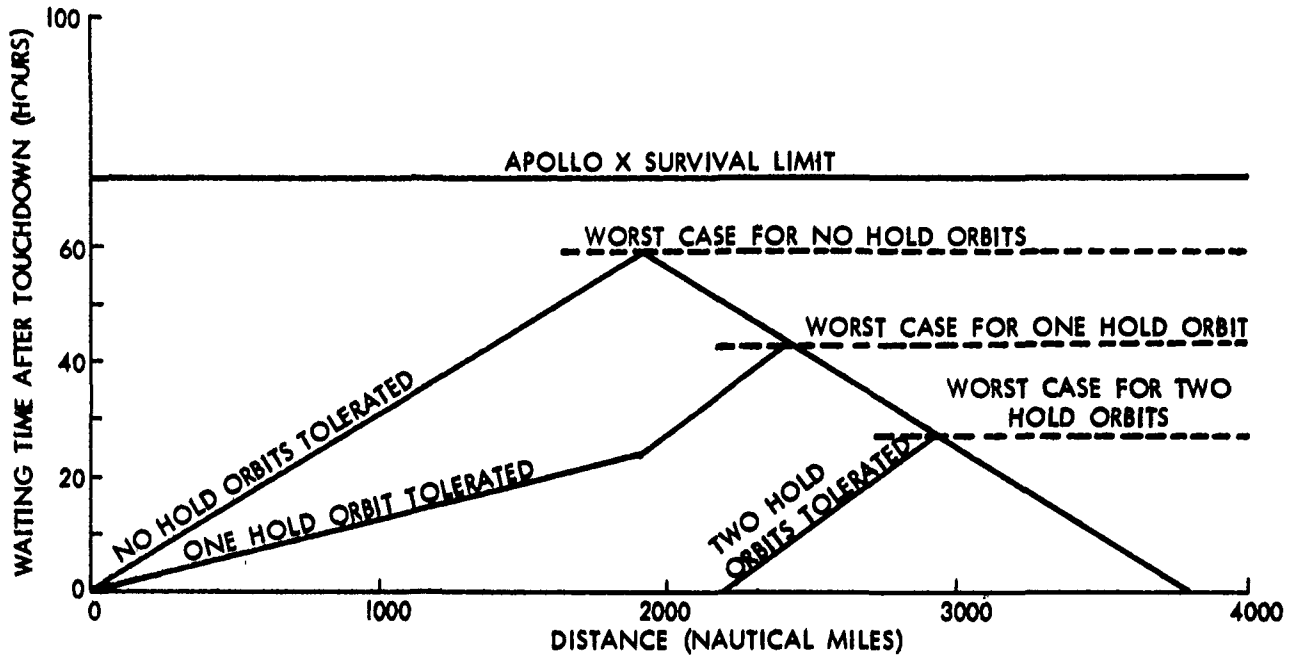

Figure 5. Total Crew Waiting Time for Four Recovery Ships Equally Spaced at 45 Degrees South Latitude

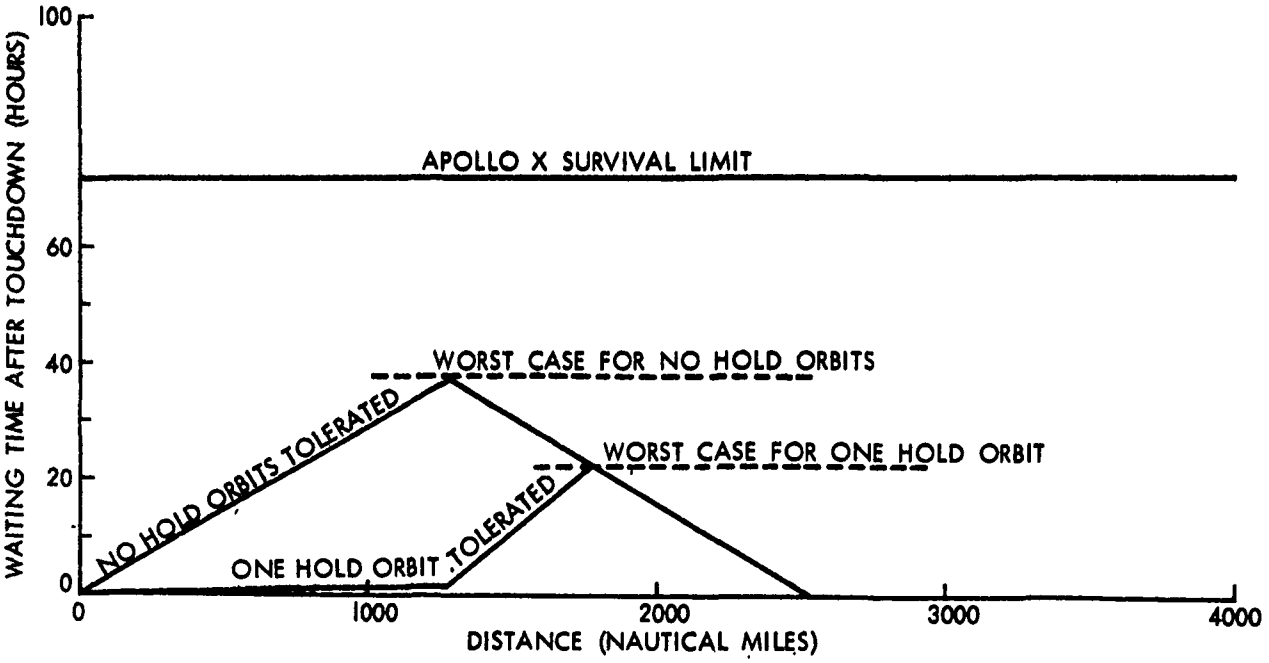

Figure 6.. Total Crew Waiting Time for Six Recovery Ships Equally Spaced at 45 Degrees South Latitude

RELIABILITY AND CONFIGURATION ANALYSES

RELIABILITY APPORTIONMENTS ANALYSIS

One of the principal study goals was to determine the maximum mission duration capability of the lunar mapping vehicle in both low-inclination and polar earth orbits. In particular, it was desired to establish the feasibility of 45-day earth-orbit missions.

It was reasonable to expect a greater mission duration capability in earth orbit (for the same mission success and crew safety probabilities) because of the following reasons:

1. Several systems have lesser reliability requirements in earth orbit because of different abort criteria.

2. Some systems (particularly the SM main propulsion) have significantly lower operating times and fewer restarts in earth-orbit missions.

3. There is no earth-orbit mission that imposes a greater performance requirement on any vehicle system than a comparable lunar-orbit mission.

4. The guidance and navigation system is not used at all for the low-inclination orbit earth missions and is used only during ascent for the polar earth-orbit missions.

Reliability apportionments for earth-orbit missions must be set to reflect the above conditions if a realistic evaluation of the capability of the lunar mapping vehicle in earth orbit is to be made.

Ideally, all system reliability apportionments for the lunar-mapping mission Apollo X should have been set early in the study so that the reliability versus time curve for each system would intersect the apportionment line at exactly 34 days. This procedure would result in the most effective system design and minimum spare requirements; however, this obviously could not be done at that time, because it presumed the existence of reliability versus time curves which were not available from the subcontractors until later in the study. Therefore, if all systems were designed and spared to meet the original nonoptimum apportionments set for the lunar-mapping vehicle, there would be some increment of total vehicle reliability built into the real vehicle that would not be apparent from the reliability data. This is true since nonoptimum apportionments always have a one-sided effect and result

in vehicle overdesign and/or oversparing if the reliability goals are met for each system. Iterative changes in the reliability apportionments, as more information became available, have resulted in more nearly optimum lunar-mission apportionments as determined by Apollo X system performance capabilities.

The reliability apportionments and reliability/time curves for the lunar mission facilitate the development of corresponding data for the earth missions. The earth-orbit curves were drawn by consideration of the variations caused by dissimilar abort conditions and system operating profiles. The earth-orbit system reliability apportionments were then determined from these curves by a very simple method involving the solution of a set of equations by a computer.

It should be noted that only selected Apollo X systems had to be considered, because the reliabilities of systems that were relatively independent of mission type, mission duration, and design factors were factored out of the required total reliability product, thereby reducing the scope of the problem. It was decided that the structure, launch escape, earth landing, and CM reaction control systems would fall in such a group. The product of the reliabilities of these systems was found to be 0.99984 from the following data:

Apollo X System	Reliability in Earth Orbit
Structure	0.99995
Launch escape	0.99999
Earth landing	0.99994
CM - RCS	0.99996

The laboratory module was assigned a reliability of 0.99585. The booster reliability was assumed (by a ground rule) to be 0.95 and GOSS and GSE reliabilities were calculated to be 0.99630 and 0.99984, respectively. Thus, the residual reliability value, R*, for the remaining nine critical Apollo X systems of interest was found to be 0.95515 by dividing the mission reliability goal of 0.90 by (0.99984 x 0.99585 x 0.95 x 0.99630 x 0.99984).

As was pointed out previously, the system apportionments for the lunar mission were not optimally set even though each system had to meet the design goal of at least a 34-day life. Now, suppose that one of the systems, say system A, had the same reliability-versus-time curve in both lunar and earth orbits and that the system just met its reliability goal for 34 days in lunar orbit. If the apportionment for this system was not

relaxed in going from the lunar to the earth mission, it is clear that system A would also limit the earth mission to 34 days, because its reliability-versus-time curve had not changed. Optimum adjustments in the apportionment of the Apollo X systems of interest (i.e., those adjustments allowing the maximum mission duration) obviously had to be determined. It should be emphasized that the real chances of mission success in earth orbit would not and could not depend on whether any apportionment adjustments were actually made since these were purely mathematical changes; however, the changes were believed to be justified because of the wide use and acceptance of reliability data and because the increased probability of success of the earth mission (as contrasted to the lunar mission) was thereby made evident by quantification.

The problem could be stated as follows. Determine a consistent set of reliability apportionments for the nine Apollo X systems of interest that would maximize the mission duration in earth orbit, T. Alternatively stated, the problem was to find a duration, T, such that each system reliability-versus-time curve intersected a consistent set of system apportionments at this value of T.

An examination of the reliability-versus-time curves for the Apollo X systems of interest showed that, for the short 11-day period (34 days to 45 days) of interest, the curves were very nearly straight lines in all cases. In other words the change in reliability from 34 to 45 days could be approximated by straight lines with negligible error.

The reliability-versus-time curves for the systems of interest in earth orbit were then inspected to determine the data presented in Table 36.

Because of the assumed linear approximations, a linear equation was written to represent the reliability-versus-time relationship for each of the nine systems when in a low-inclination earth orbit of duration T_L:

$$(R_{1,L} - R_{1,34,L}) = \frac{\Delta R_{1,L}}{11} (T_L - 34)$$

. .

. .

. .

. .

$$(R_{9,L} - R_{9,34,L}) = \frac{\Delta R_{9,L}}{11} (T_L - 34)$$

Table 36. Data Table for Calculation of Optimum System Reliability Apportionments for Apollo X Earth Missions

System Name	System Index Number	Low-Inclination Earth Orbit		Polar Earth Orbit	
		Reliability at 34 Days ($R_{i,34,L}$)	Probability Change Between 34 and 45 Days ($\Delta R_{i,L}$)	Reliability at 34 Days ($R_{i,34,P}$)	Reliability Change Between 34 and 45 Days ($\Delta R_{i,P}$)
SM reaction control	1	0.99990	-0.00010	0.99900	-0.00030
Environmental control	2	0.99280	-0.00380	0.99280	-0.00380
SM propulsion	3	0.99996	-0.00002	0.99996	-0.00002
Stabilization and control	4	0.99388	-0.00000*	0.99908	-0.00023
Guidance and navigation	5	1.00000	-0.00000	0 99690	-0.00040
Communications and displays	6	0.99770	-0.00130	0.99770	-0.00130
Electrical power	7	0.98400	-0.00500	0.98400	-0.00500
Cryogenic storage	8	0.99960	-0.00080	0.99960	-0.00080
Instrumentation	9	0.99940	-0.00040	0.99940	-0.00040

*S&C system is not operating between 34 and 45 days.

Also, the nine reliability apportionments had to equal the residual reliability value R* when multiplied together.

Hence

$$R_1 \times R_2 \times \ldots \times R_9 = 0.95515$$

Thus, there was formulated a set of ten equations in the ten unknowns, $R_{1,L}$, $R_{2,L}$, . . . , $R_{9,L}$, T_L which were solved by a computer program to find the optimal reliability apportionments $R_{1,L}$, . . . , $R_{9,L}$, as well as the maximum mission duration in earth orbit, T_L, as set by reliability considerations. Similarly, the set of ten equations involving $R_{1,P}$, $\Delta R_{1,P}$, and T_P were solved, using the data for the polar earth orbit. The results showed that, based solely on reliability considerations, the lunar mapping vehicle systems were capable of a 46-day mission in both low-inclination and polar earth orbits. Since consumables were sized for a maximum 45-day mission in each case, the reliability apportionments for 45 days were of interest and are included in Table 37. The functional relationships are depicted in Figure 7

Because the lunar mapping vehicle systems were shown to be adequate in all cases for either low-inclination or polar earth orbits of 45 days duration, it was concluded that system location and arrangement were the primary factors to be considered in the tradeoff analysis.

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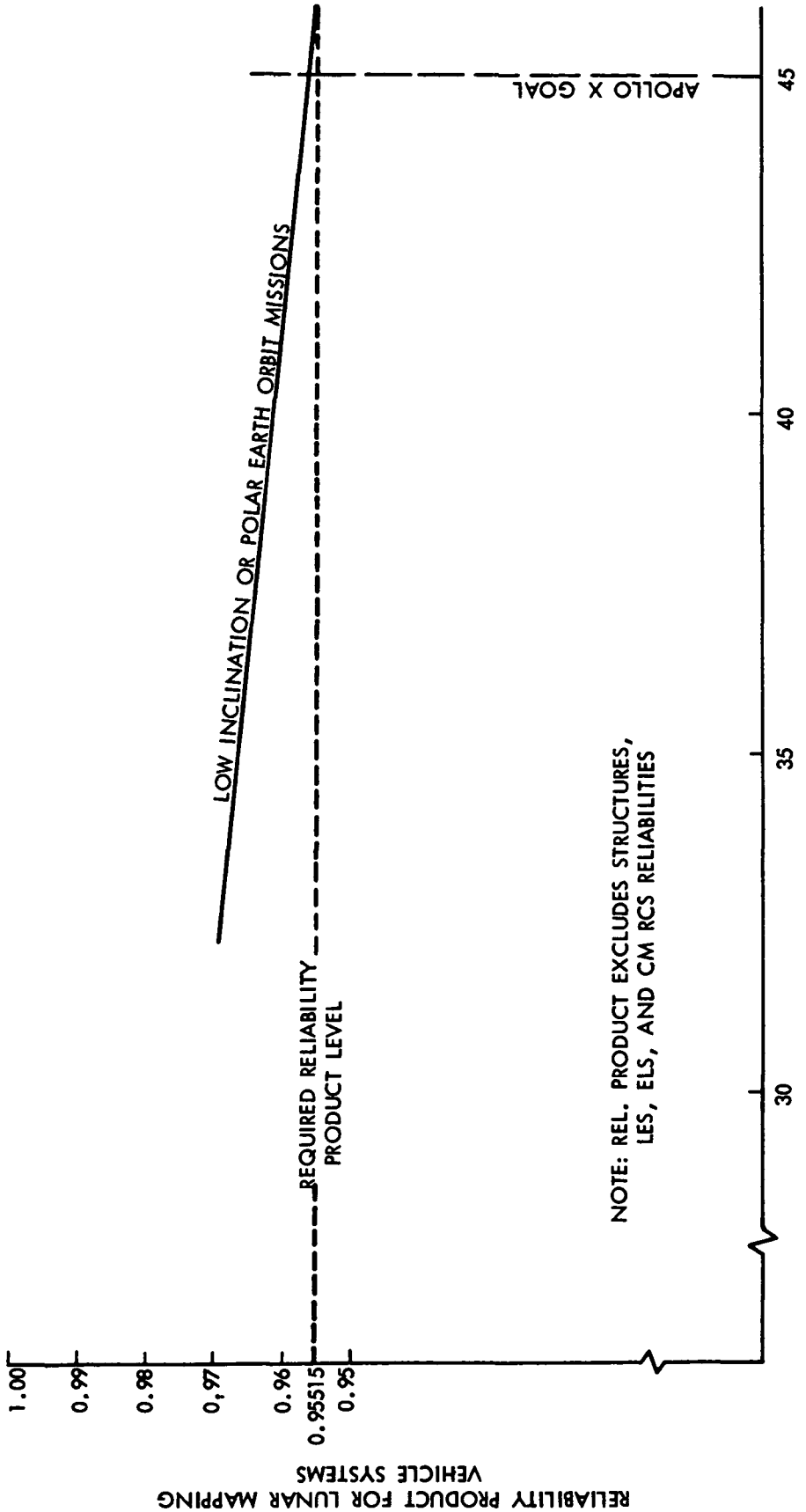

Figure 7. Reliability Apportionment Analysis

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CONFIGURATION ANALYSIS

A summary of the systems and structure making up the recommended configurations for missions 1A, 1B, 2A, 2B, 3, and the lunar mapping mission is presented in Table 38. In most cases the changes are either self-explanatory or have been adequately described in other sections of this volume. The rationale for the choices is discussed briefly in this section.

LIFE SUPPORT AND CREW SYSTEM

The right-hand seat is removed from the two-man configuration because this allows for the most efficient use of the extra space. Because working space in the command module is extremely limited in the two-man configuration, this fact overrides the attenuator redesign problem incurred by nonsymmetrical seat loading.

Two portable life support systems are stored in the command module for all missions. These systems are required for extravehicular activities on all earth-orbiting missions (one man in space, one man available for rescue) and for safe crew return in case of an emergency during the lunar mapping mission.

EXPERIMENTAL SYSTEM

The equipment and supplies required by the various experiment packages are documented in Volume 3 of this report.

ENVIRONMENTAL CONTROL SYSTEM

A two-gas atmosphere is indicated in order to eliminate possible undesirable physiological effects on the crew caused by long exposure to pure oxygen. A 27.5-inch sphere in the center section of the service module contains sufficient cryogenic nitrogen for three men for 45 days and hence is installed in all configurations of the standardized vehicle.

Trace contaminants in the atmosphere may be a serious problem on long duration missions. Catalytic burners, air filters, and ultraviolet lamps are added to the environmental control system in all concepts to alleviate this danger.

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Table 38. Systems for Apollo X Conceptual Missions

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
Life support and crew	Right-hand crew couch removed and earth landing shock attenuator system redesigned Personal hygiene, waste management and food supplies (for 90 man-days) stored in CM Two space suits, pressure helmets, and communication helmets Two PLSS in CM	Personal hygiene, waste management and food supplied (for 135 man days) stored in lab module Two PLSS in CM	Personal hygiene, waste management and food supplies (for 135 man-days) stored in laboratory module Two PLSS in CM	Personal hygiene, waste management and food supplies (for 135 man-days) stored in laboratory module Transearth life support requirements (9 man-days) transferred to CM before laboratory module separation Two (or three) PLSS in CM
Experimental	Experimental equipment stored in CM module	Experimental packages stored in lab module	Experimental packages stored in lab module	Spectrozonal and cartographic cameras installed in laboratory module

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
Experimental (Cont)	Natural science test equipment stored on boom attached to SM	Natural science test equipment stored on boom attached to SM Data and equipment to be returned to earth are stored in CM before lab module separation	Telescope installed in lab module (mission 2B only) Spectrozonal and cartographic cameras installed in laboratory module (mission 3 only) Natural science test equipment on boom attached to SM Data and equipment to be returned to earth are stored in CM before laboratory module separation	Film is stored in CM before laboratory separation
ECS	60 Apollo-type LiOH canisters stored in CM (or store four	Four Apollo-type LiOH canisters in CM	Four Apollo-type LiOH canisters stored in CM	14 Apollo-type LiOH canisters stored in CM

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
ECS (Cont)	Apollo-type canisters in CM and add three Gemini-type canisters to sector 3 of SM) Add two-gas atmosphere controls and displays in CM Trace contaminant removal system installed in CM Add 27.5-inch (outside diameter) N_2 sphere in center sector of SM One 40-inch sphere for supercritical ECS (and EPS) oxygen in each of sectors 1 and 4 of SM	Trace contaminant removal system installed in CM Duct and blower for air return from laboratory module Add 27.5-inch (outside diameter) N_2 sphere in center sector of SM (Two spheres for up to 10 CM repressurizations) 86 Apollo-type LiOH canisters stored in laboratory module One 40-inch sphere for supercritical ECS (and EPS) oxygen in each of sectors 1 and 4 of SM Add glycol loop to lab module	Trace contaminant removal system installed in CM Duct and blower for air return from laboratory module Add 27.5-inch (outside diameter) N_2 sphere in center sector of SM (Two spheres for up to 10 CM/laboratory repressurizations) 86 Apollo-type LiOH canisters stored in laboratory module One 40-inch sphere for supercritical ECS (and EPS) oxygen in each of sectors 1 and 4 of SM Add glycol loop to lab module	Trace contaminant removal system installed in CM Duct and blower for air return from laboratory module Add 27.5-inch (outside diameter) N_2 sphere in center sector of SM 54 Apollo-type LiOH canisters stored in laboratory module One 40-inch sphere for supercritical ECS (and EPS) oxygen in each of sectors 1 and 4 of SM Add glycol loop to lab module

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
Electric power	Add two Apollo fuel cells to sector 1 of SM One 40-inch sphere for supercritical EPS (and ECS) O_2 in each of sectors 1 and 4 of SM One 40-inch x 58-inch tank for supercritical EPS H_2 in each of sectors 1 and 4 of SM	Add two Apollo fuel cells to sector 1 of SM One 40-inch sphere for supercritical EPS (and ECS) O_2 in each of sectors 1 and 4 of SM One 40-inch x 58-inch tank for supercritical EPS H_2 in each of sectors 1 and 4 of SM	Add two Apollo fuel cells to sector 1 of SM One 40-inch sphere for supercritical EPS (and ECS) O_2 in each of sectors 1 and 4 of SM One 40-inch x 58-inch tank for supercritical EPS H_2 in each of sectors 1 and 4 of SM	Add two Apollo fuel cells to sector 1 of SM One 40-inch sphere for supercritical EPS (and ECS) O_2 in each of sectors 1 and 4 of SM One 40-inch x 58-inch tank for supercritical EPS H_2 in each of sectors 1 and 4 of SM
Guidance and Navigation	Block II G&N system (except optics) removed GOSS data for navigation	Block II G&N system (except optics) removed GOSS data for navigation	Block II G&N with 102 pounds of spares added to increase reliability of system GOSS data for navigation	Block II G&N with 102 pounds of spares added to increase reliability of system GOSS data for navigation

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
SCS and RCS	Add booster monitor capability	Add boost monitor capability	Replace Block II RCS propellant and oxidizer tanks in sectors 2,3,5, and 6 of SM with LEM tankage Provide two helium spheres for RCS pressurization Active thermal control on RCS thrusters	Add horizon sensors Add 47 pounds of spares Replace Block II RCS propellant and oxidizer tanks in sectors 2,3,5, and 6 of SM with LEM tankage Provide two helium spheres for RCS pressurization Active thermal control on RCS thrusters
Communication and displays	Add two C-band transponders	Add two C-band transponders Install intercom to laboratory module Remove rendezvous radar from SM	Add two C-band transponders Install intercom to laboratory module	Add two C-band transponders Install intercom to laboratory module Remove rendezvous radar from SM

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
	Move right-hand display panel for center couch visibility Remove rendezvous radar from SM Remove high-gain S-band antenna from SM	Remove high-gain S-band antenna from SM	Remove rendezvous radar from SM Remove high-gain S-band antenna from SM	
Structures (Secondary)	Remove block II docking hardware Modify aft bulkhead, side panels, supports, and access doors in SM	Modify aft bulkhead, side panels, supports, and access doors in SM Laboratory module sized to fit within LEM adapter	Modify aft bulkhead side panels, supports, and access doors in SM Laboratory module sized to fit within LEM adapter	Modify aft bulkhead, side panels, supports, and access doors in SM Laboratory module sized to fit within LEM adapter

Table 38. Systems for Apollo X Conceptual Missions (Cont)

System	Two-Man Earth Mission 1A	Three-Man Earth Missions 1B and 2A	Three-Man Earth Missions 2B and 3	Three-Man Lunar Mapping Mission
		Docking ports on each end of laboratory module	Docking ports on each end of laboratory module	Docking ports on each end of laboratory module
Propulsion	Block II SPS fuel tanks replaced with one 45-inch sphere in each of sectors 3 and 6 of SM Block II SPS oxidizer tanks replaced with one 51-inch sphere in each of sectors 2 and 5 of SM One 42-inch sphere for helium in center sector of CM	Same as for two-man earth mission 1A	Same as for two-man earth mission 1A	No changes in Block II system

Two 40-inch spheres located in sectors 1 and 4 of the service module can supply the cryogenic oxygen requirements of both crew (three men) and fuel cells for 45 days; hence, these spheres are installed in all configurations of the standardized vehicle.

In the vehicle configurations with a laboratory module, it is always necessary to store enough Apollo LiOH canisters in the command module to supply the crew after laboratory module separation. Four canisters are required for earth orbit and 14 canisters are required for the lunar mapping missions. The remaining canisters are stored in the laboratory module.

For the two-man configuration, all 60 of the required Apollo canisters will be stored in the command module if volume permits. If working room is unduly restricted by this arrangement, three Gemini-type LiOH canisters can be installed in the forward region of sector 3 of the service module.

ELECTRIC POWER SYSTEM

In order to obtain an adequate reliability level for the electric power system, it is necessary to install two additional fuel cells in sector 1 of the service module. These cells are additional backup to the one Block II spare cell and will not be started until cell failures require it.

Two 40-inch x 58-inch cylindrical tanks located in sectors 1 and 4 of the service module can meet the cryogenic hydrogen requirements of two fuel cells supplying the expected power demands over a 45-day mission and are thus adequate for all configurations of the standardized vehicle. The reactant oxygen is shared with the metabolic O_2 supply as described in the preceding section.

GUIDANCE AND NAVIGATION SYSTEM

The guidance and navigation system (except optics) is removed for earth missions 1A, 1B, and 2A, because GOSS can supply navigational data and because orientation requirements are minimal. Astronomical observations (mission 2B) and earth mapping (mission 3) involve attitude hold and positional requirements that can be met either by restoring the G&N system or by installing a special computer. Additional studies will be needed to show the advantages of each of these methods. A spares package weighing 102 pounds must be included with the G&N system to meet the reliability apportionment.

The lunar mapping operation can be performed satisfactorily with a modified stabilization and control system. However, certain spacecraft

maneuvers connected with reaching and returning from the moon have a much greater margin of safety when a G&N system is on-board. For this reason, the system is made part of the lunar mapping configuration.

STABILIZATION AND CONTROL SYSTEM -- REACTION CONTROL SYSTEM

The stabilization and control system is modified to obtain booster monitor capability in missions 1A, 1B, and 2A, because the guidance and navigation system is removed for these missions. This modification involves relatively minor electronic and logic changes. The G&N optical system is also incorporated into the stabilization and control system to meet guidance and control requirements in connection with the use of navigational data from GOSS.

The addition of a horizon sensor and 47 pounds of spares to the SCS allows lunar mapping to be accomplished without operating the guidance and navigation system. This addition in effect allows the G&N system to meet its reliability apportionment.

Configurations for missions 2B, 3, and the lunar mapping mission include active thermal control for the RCS thrusters to prevent excessive propellant heating and cooling when fixed orientation relative to the sun is maintained for long periods. An analysis showed that the weight penalty incurred by this method of thermal control is much less than the penalty that results from spacecraft rotation and a separate camera mounting.

COMMUNICATIONS AND DISPLAYS

The modifications to the communications system consist primarily of the addition of two (one redundant) C-band transponders to increase the reliable lifetime of the system. Most of the communications system will be in use or on standby at all times throughout the mission, because reliability may actually be decreased by frequent shutdowns.

The high-gain S-band is removed from the service module on the earth mission configurations, because it is not required for earth-orbital ranges. The Apollo rendezvous radar is not used and is removed from the service module.

An intercom is installed between the command module and the laboratory module to maintain communication.

The Apollo displays remain in the command module, however, the right-hand display panel must be moved in the two-man configuration

(mission 1A) to allow visibility from the center couch. Special displays for the experiments are described in Volume 3 of this report.

STRUCTURES

Modifications to the command module are limited to the removal of the unused docking hardware from the two-man configuration.

The modifications to the service module, although quite numerous, involve only secondary structural changes to the aft bulkhead, side panels, supports and access doors to accommodate the additional fuel cells, enlarged cryogenic tanks, nitrogen spheres, and LEM RCS propellant and oxidizer tanks. These changes to the service module structure are designed to make modifications essential in the earlier missions compatible with the later lunar vehicle configuration.

The laboratory module is sized to fit inside the LEM adapter (using the present LEM attach points). The remaining space in the adapter is left for either another identical laboratory module or the LEM descent stage engine. This approach provides maximum versatility for future missions without seriously reducing the mission effectiveness of the Apollo X.

PROPULSION

Because an earth mission does not require nearly as much fuel and oxidizer as a lunar mission, the Block II SPS tanks in these configurations are replaced with two 45-inch spheres for fuel in sectors 3 and 6 and with two 51-inch spheres for oxidizer in sectors 2 and 5. One 42-inch sphere for helium in the center sector of the command module is adequate for all earth missions.

There are no changes in Block II SPS for the lunar mapping mission.

The service module main engine has adequate performance capability and reliability for all missions.

PERFORMANCE ANALYSIS

The objective of the performance analysis effort expended during this study was to establish representative Apollo X mission profiles, with their concomitant environments and operating requirements, that would provide a basis for meaningful subsystem studies. Useful payload weight characteristics for these missions also were determined to aid in evaluating various subsystem approaches to life extension

To satisfy the above objectives, earth-orbital missions at low and high inclinations and lunar missions were synthesized The range of payload capabilities required for these missions necessitated examination of both the Saturn-IB and the Saturn-V launch vehicles. The lunar orbit mission was found to be critical for a number of subsystems, and the abort characteristics for the polar lunar orbit were found to have significant effects on allowable payload.

EARTH-ORBITAL MISSIONS

The investigation of launch vehicle performance was directed toward refining data obtained during the recent Extended-Mission Apollo study and toward defining off-normal capabilities involved in obtaining increased inclination with various end boost configurations Preliminary analyses indicated that the Saturn-IB launch vehicle, which was considered in the previous studies, would not provide sufficient capability to achieve a polar orbit from the Atlantic Missile Range (AMR). For this reason, the Saturn-V booster was selected to provide the required payload weight at the higher inclinations.

SATURN IB

A summary of the Saturn-IB performance capabilities, is shown in Figure 8. These data were established by using the NASA/MSFC guaranteed payload of 32,500 pounds to a 105 nautical mile, 33.5-degree inclined orbit. With the stage data that matches this performance, the payload variation with orbit altitude and launch azimuth was determined. For the mission altitude of 200 nautical miles, the booster payload capability shown for a due east launch is approximately 22,500 pounds. If the projected uprating is accomplished, the latter payload value would be increased to about 27,500 pounds. The design approach for the Apollo X is illustrated by

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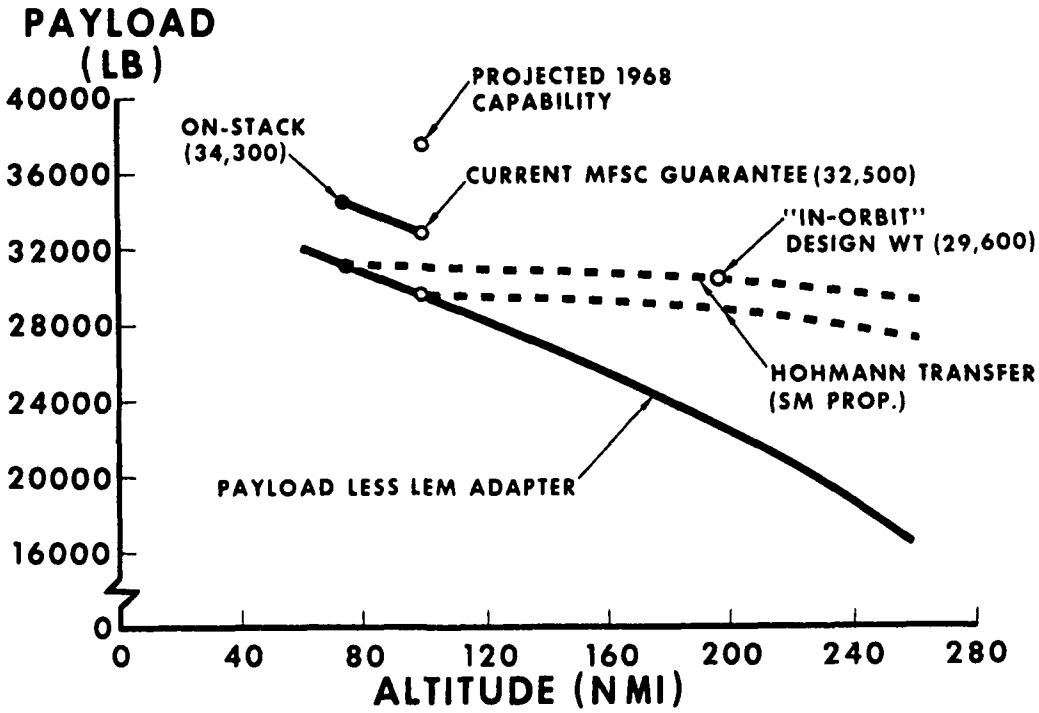

Figure 8. Saturn IB Performance

the dashed lines in Figure 8 and is derived by using the Apollo service propulsion system (SPS) to achieve the mission orbit following a S-1B boost to circular orbit at 80 or 105 nautical miles. Design payload weight capability using this approach would be 29,600 pounds, exclusive of the LEM adapter. The projected 1968 capability of the uprated S-IB using this same approach could be about 33,000 pounds, which would allow a considerable growth from the design value

Time history variations of basic flight parameters during the S-IB ascent trajectories previously mentioned are presented in Figures 9 and 10 for the 80- and 105-nautical mile burnout altitudes, respectively The S-IB first stage thrust programming calls for shutdown of the outboard engines prior to burnout to decrease the severity of transients, and main engine cutoff (MECO-1) occurs at 153 seconds. Following a nominal 5.5-second coast, the S-IB stage is jettisoned and the S-IVB is started At 168.5 seconds, with the S-IVB stage at full thrust, the launch escape system is jettisoned, and the vehicle is steered on an optimal path to S-IVB main engine cutoff (MECO-2) at 622.5 seconds Comparison of the recommended ascent to 80 nautical miles with the MSFC standard 105-nautical-mile trajectory reveals a maximum dynamic pressure increase from 553 psf to 566 psf and 4 seconds less S-IVB burn time.

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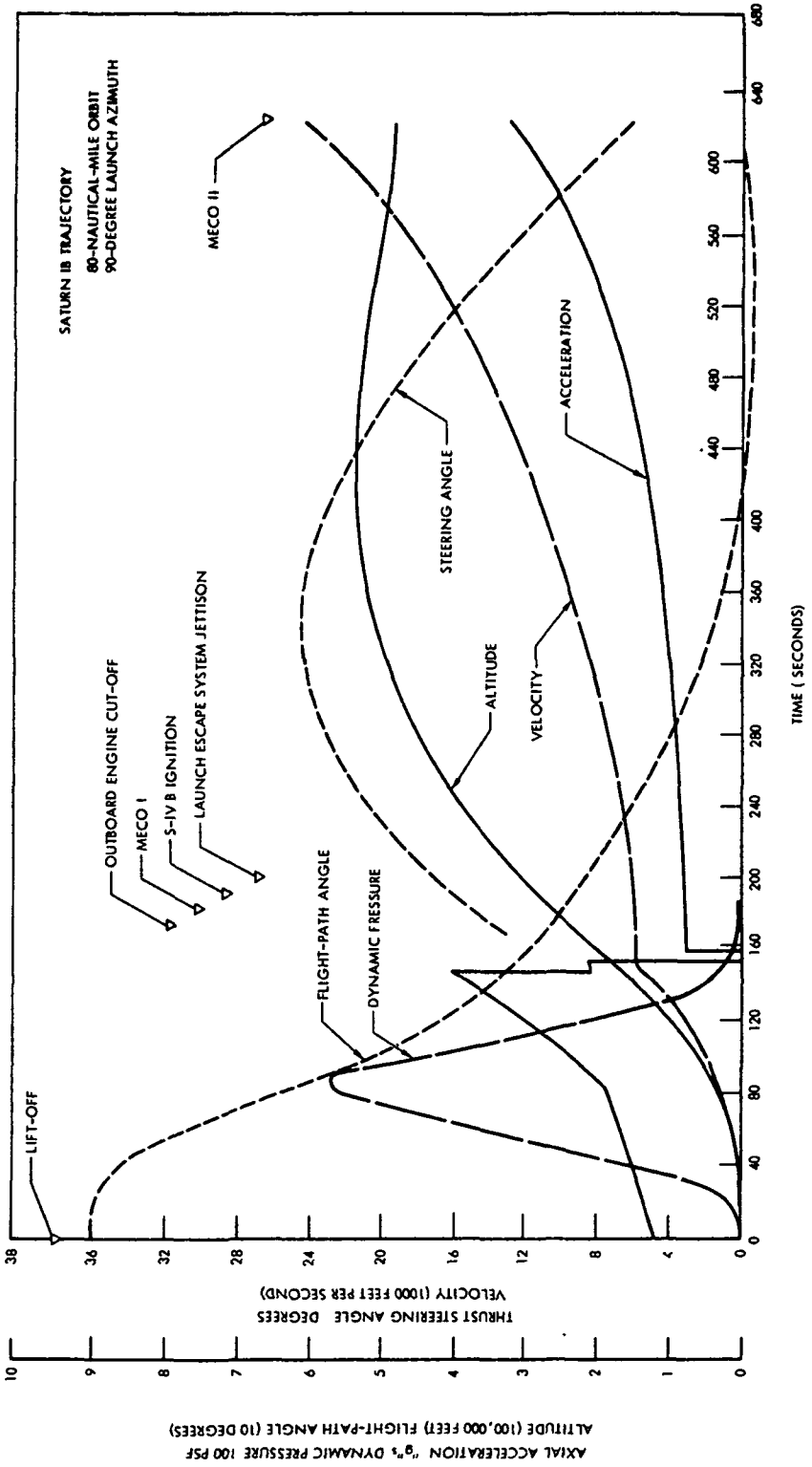

Figure 9. S-IB Trajectory, Due East Launch

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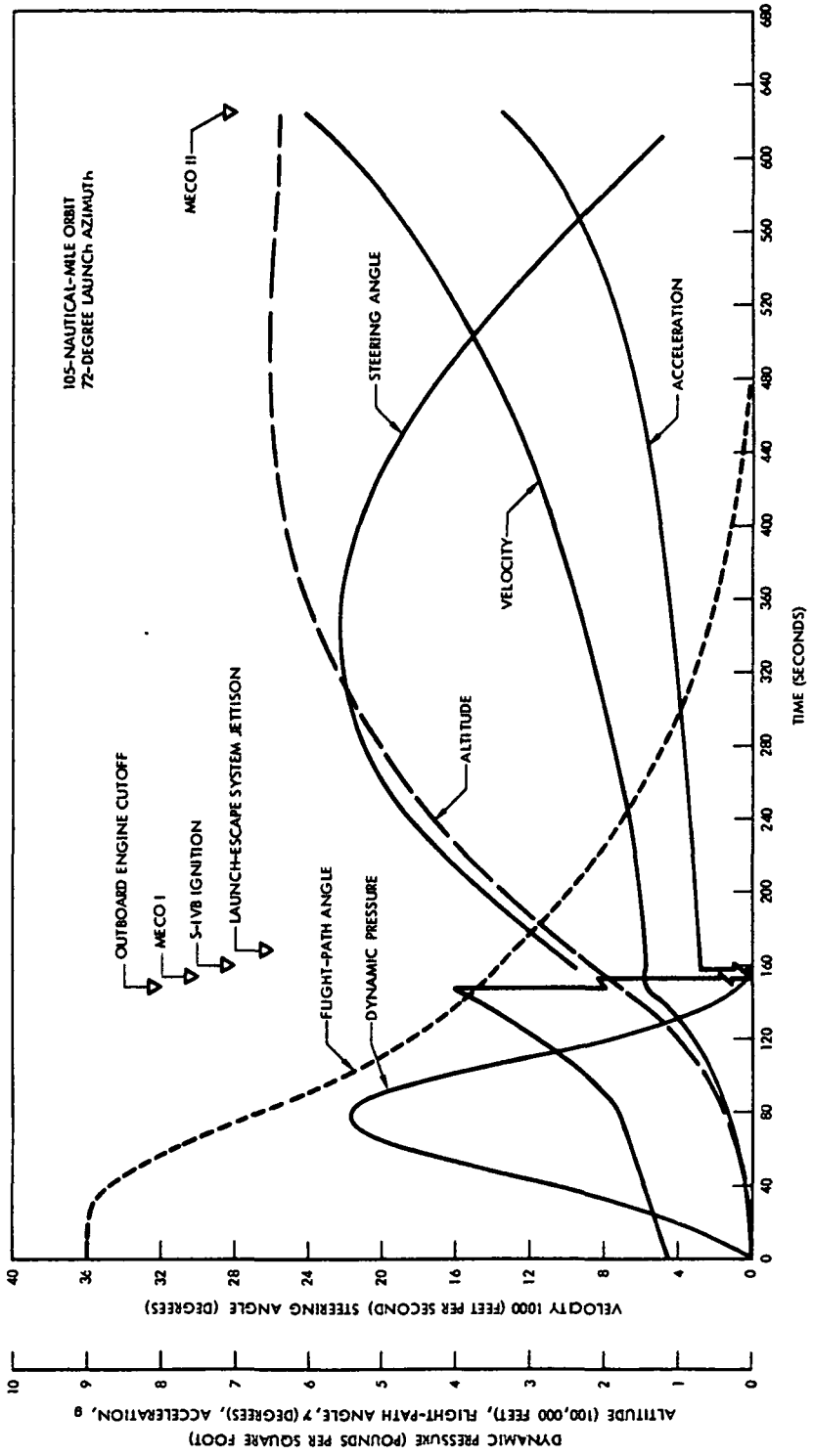

Figure 10. Standard Trajectory

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Alternative inclination capability for the Saturn-IB launched Apollo X is presented in Figure 11. The performance given by the solid curve in this figure results from increasing the inclination by changing the initial launch azimuth. Further increases can be made after orbit has been established. This approach, illustrated by the dashed curve in Figure 11, is costly and results in large decreases in payload capability. Review of these data indicate that with orbit payloads of interest in the present study only moderate inclinations can be achieved with the current Saturn-IB launched system. For this reason, orbital performance of the Saturn-V launch vehicle was examined.

SATURN V

The approach taken in obtaining the Saturn-V performance was similar to that used for the Saturn IB, in that the NASA/MSFC standard trajectory data was matched by using the latest available stage information. Since these studies were completed, preliminary data of revised Saturn-V capability has been received. This performance would be superior to that quoted herein. The present data is conservative by about 4000 pounds. In addition to the complete Saturn-V vehicle, the capability of a boost vehicle comprising only the S-IC and S-II stages also was established. Some of these data in terms of orbit altitude-payload variation are presented in Figure 12. Altitude variation for the two-stage vehicle would be quite similar to the curve data shown. A time history of the flight parameters for an ascent trajectory representative of this performance is shown in Figures 13 and 14. S-IC stage cutoff occurs following a 150-second burning period A coast and separation follows with S-II stage ignition at about 154.5 seconds, followed 20 seconds later by jettisoning the launch escape system. The S-II stage cutoff occurs at 543 seconds and is followed by S-II jettison and S-IVB stage ignition. Circular orbit conditions are achieved at approximately 709 seconds, and the S-IVB stage is shut down. The length of this last burning period will depend upon the initial launch azimuth and the orbit altitude desired.

The most important aspect of the Saturn V used in the current application, would be its use in providing high-inclination capabilities for spacecraft launched from the Atlantic Missile Range. The data shown in Figure 15 result when spacecraft inclinations are obtained from launch vehicle azimuth according to the geometrical relationships:

$$\text{inclination} = \arccos(\cos \text{latitude} \cdot \sin \text{azimuth}),$$

The Saturn V inclination characteristics are shown in Figure 16. These data present launch vehicle payload variation with orbit inclination for an AMR launch. The two-stage vehicle previously mentioned is shown again for comparison. The data shown by the solid curve result when inclination can be obtained by the above method.

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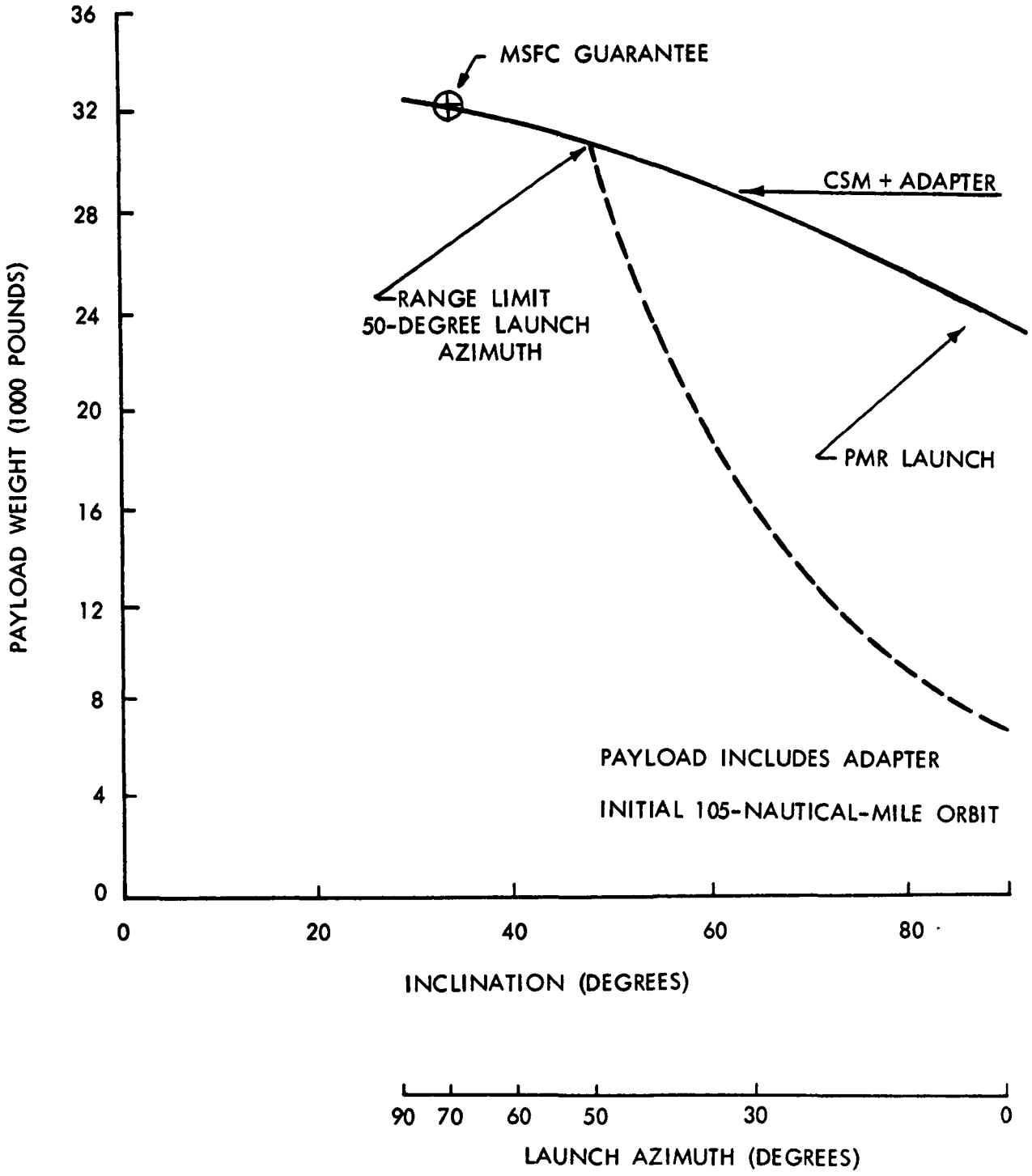

Figure 11 S-IB Payload/Inclination Summary

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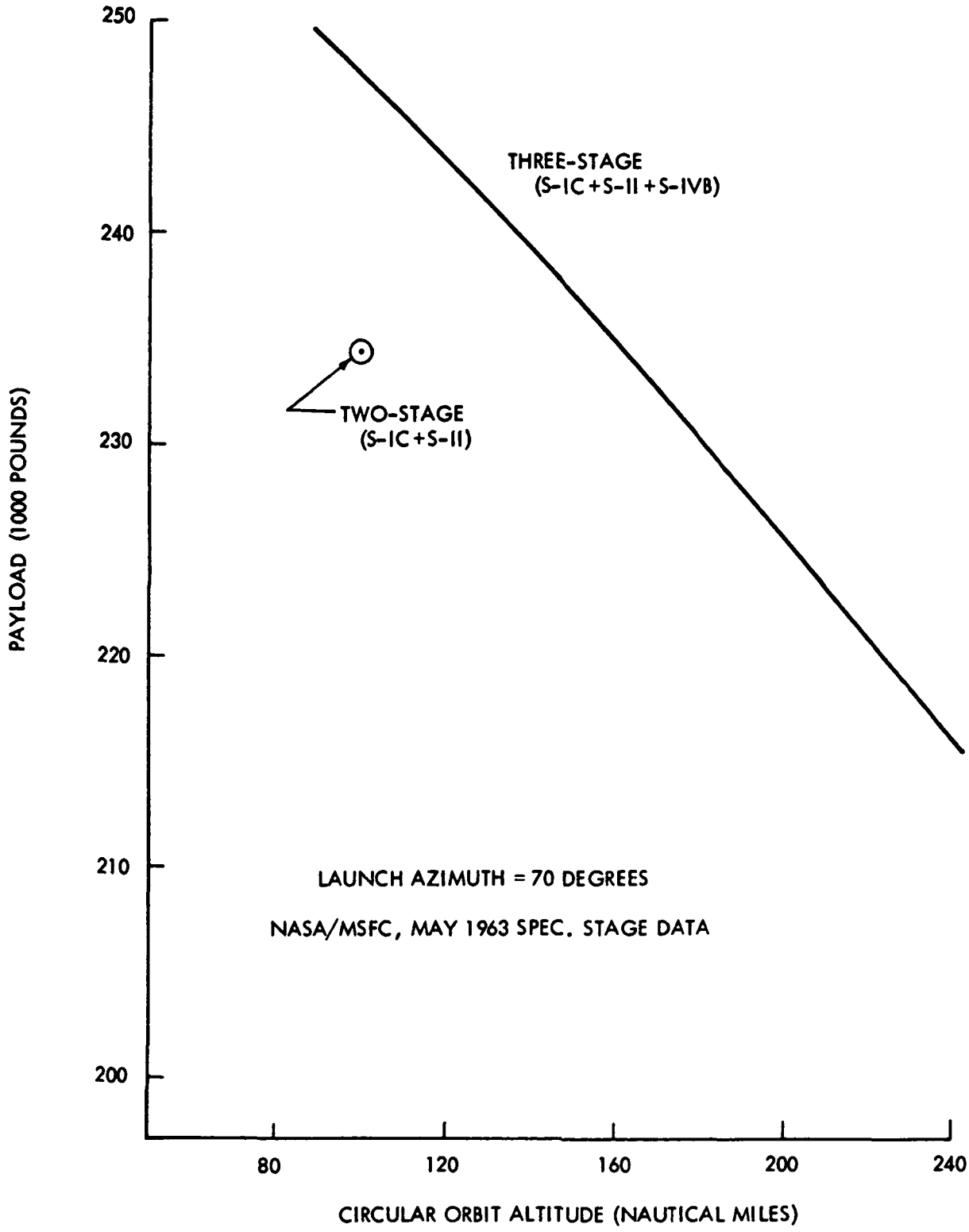

Figure 12. Saturn V Performance

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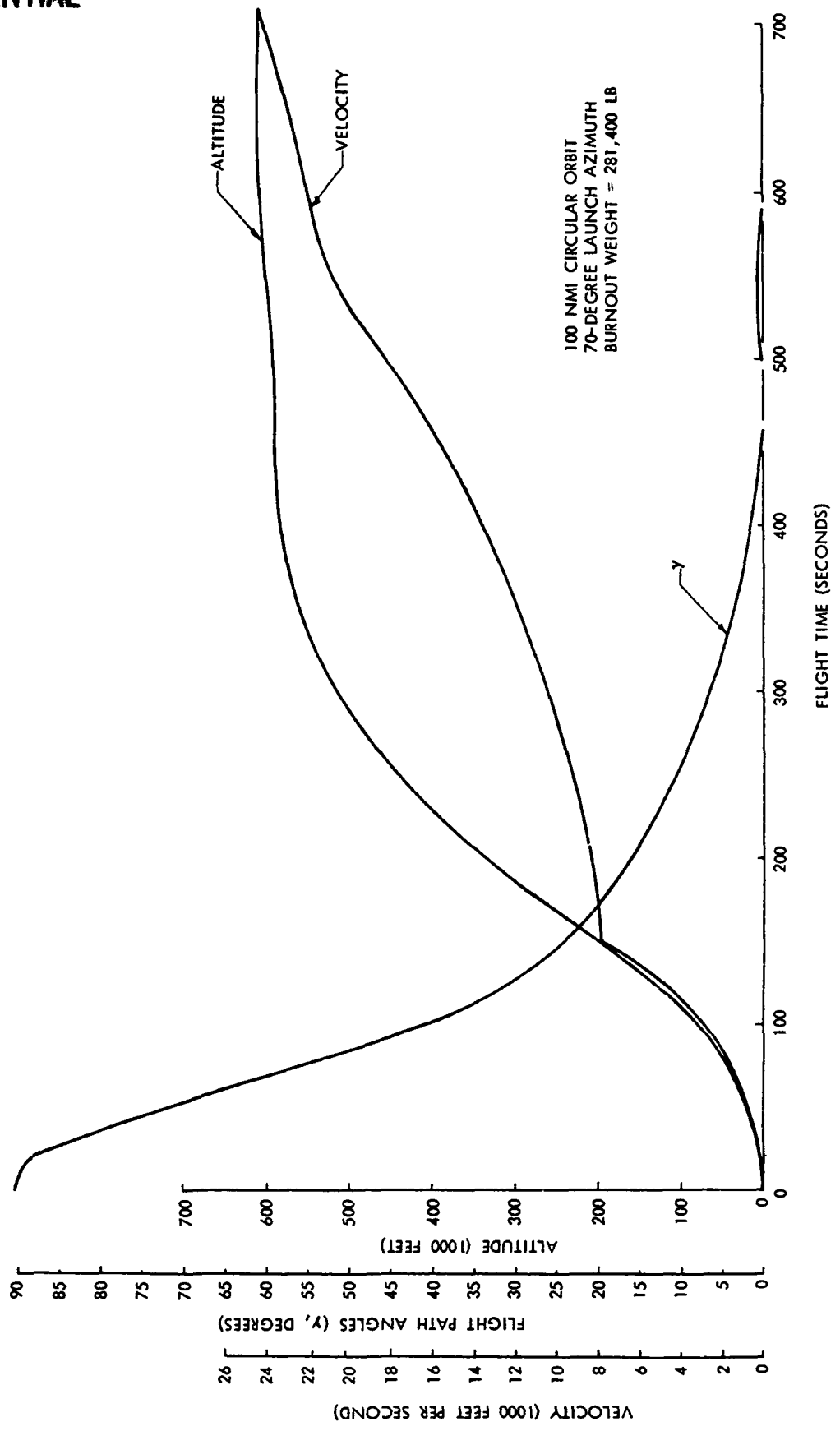

Figure 13. Saturn V Standard Trajectory

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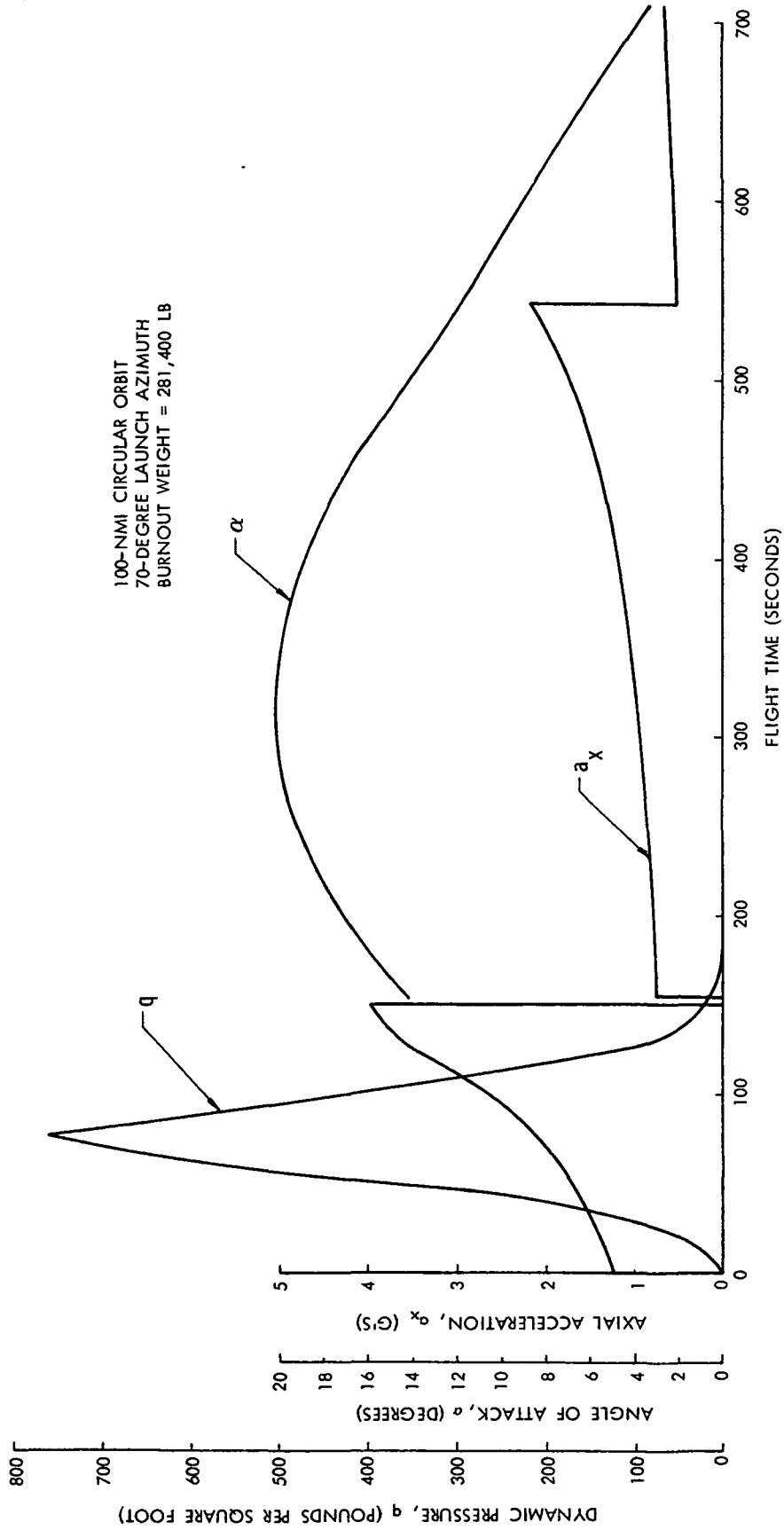

Figure 14. Saturn V Standard Trajectory

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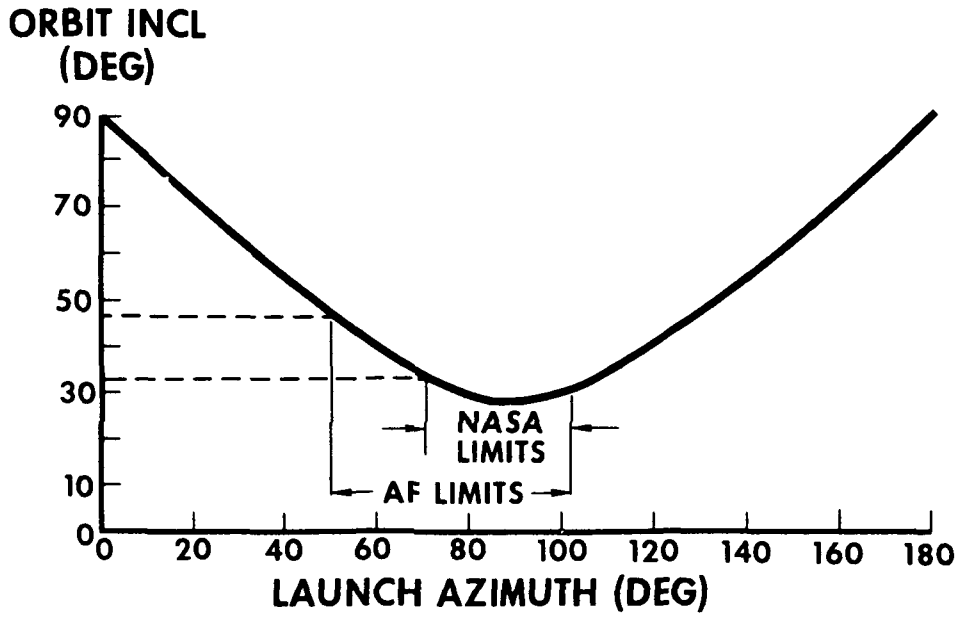

Figure 15. Launch/Orbital Characteristics

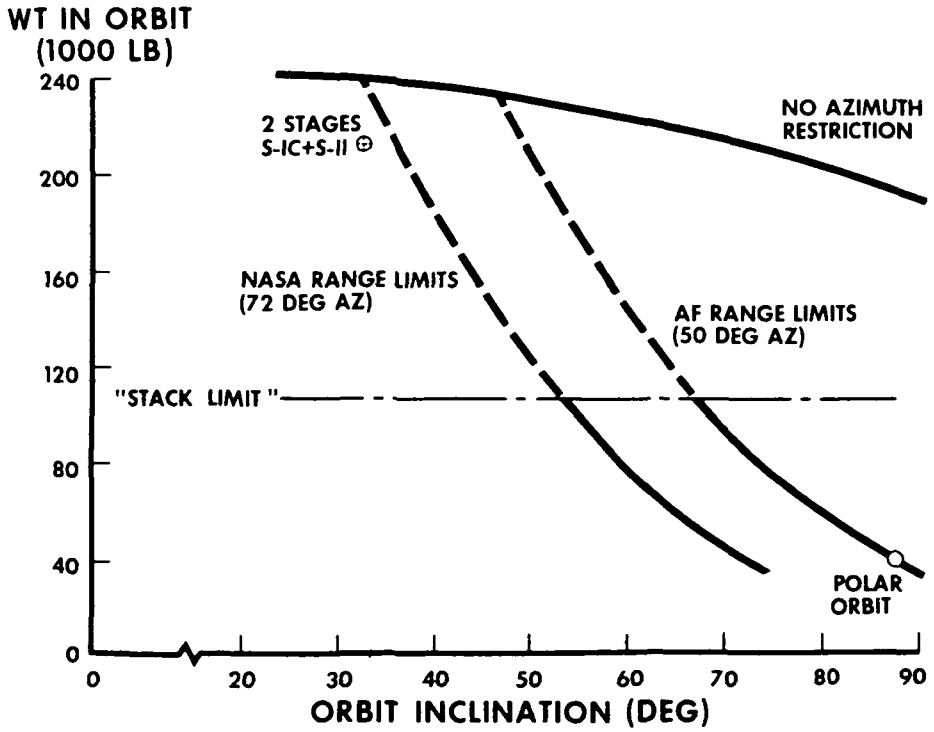

Figure 16. Saturn V Performance

The dashed curves in Figure 16 correspond to the initial launch azimuths of 72 and 50 degrees that are the NASA and Air Force Northern most range safety limits for the AMR. The payload variation results from using the Saturn-IVB stage propellant remaining after achieving circular orbit to make an orbital plane change. It should be noted that all payload weight in excess of about 100,000 pounds must be S-IVB stage propellant because of booster design factors; thus, payload values greater than 100,000 pounds could only be utilized for maneuvering purposes.

EARTH-ORBITAL CHARACTERISTICS

To define subsystem operating conditions and duty cycles, typical Apollo X flight profiles have been synthesized for low-inclination (28.5 degrees) and polar earth orbits. As previously mentioned, the launch vehicles chosen to produce these two operating conditions are different because of the large additional maneuver requirements associated with the polar orientation. The mission orbit altitude of about 200 nautical miles, chosen in previous Extended-Mission Apollo studies, has been retained as a favorable compromise between boost payload and orbit lifetime. Orbit decay characteristics associated with this altitude, shown for the range of Apollo X ballistic parameters in Figure 17, indicate that station-keeping maneuvers will not be required for the configurations and orbit time periods being considered.

A representative ascent flight profile for the low-inclination orbit mission is presented in Figure 18. The Saturn-IB sequence was outlined previously for achieving a 28.5-degree inclined orbit at an altitude of 80 nautical miles. This sequence would be followed by an orbital coast, during which time a transposition and docking maneuver would take place in the case of configurations having a laboratory module. This coast is terminated with a maneuver that changes the apogee altitude to 200 nautical miles; and approximately one-half orbit later, at the 200-nautical-mile apogee point, another maneuver circularizes the orbit. These orbital maneuvers are accomplished by the SPS. Figure 19 depicts a typical recovery flight profile that would follow the mission completion and is representative of the data for all earth-orbital concepts. The retrograde velocity increment of about 550 feet per second is provided by the SPS. After a descending orbit coast, during which time navigation and reentry corrections are accomplished, the service module is separated, and the command module commences atmospheric entry. A time history of flight parameter variations during a nominal atmospheric entry is shown in Figure 20 These data assume a 45-degree roll orientation to account for crossrange maneuver effects.

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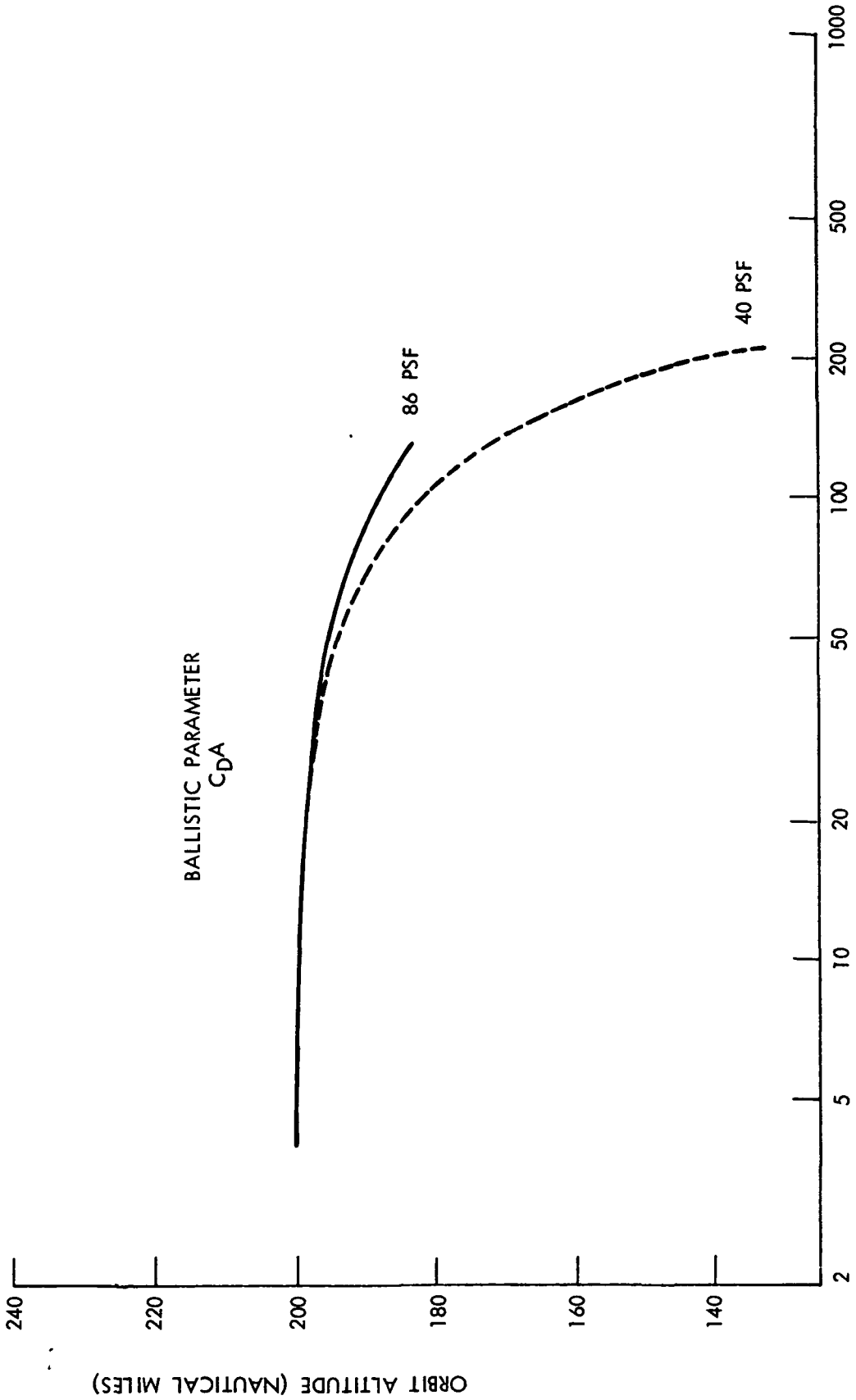

Figure 17. Orbit Decay

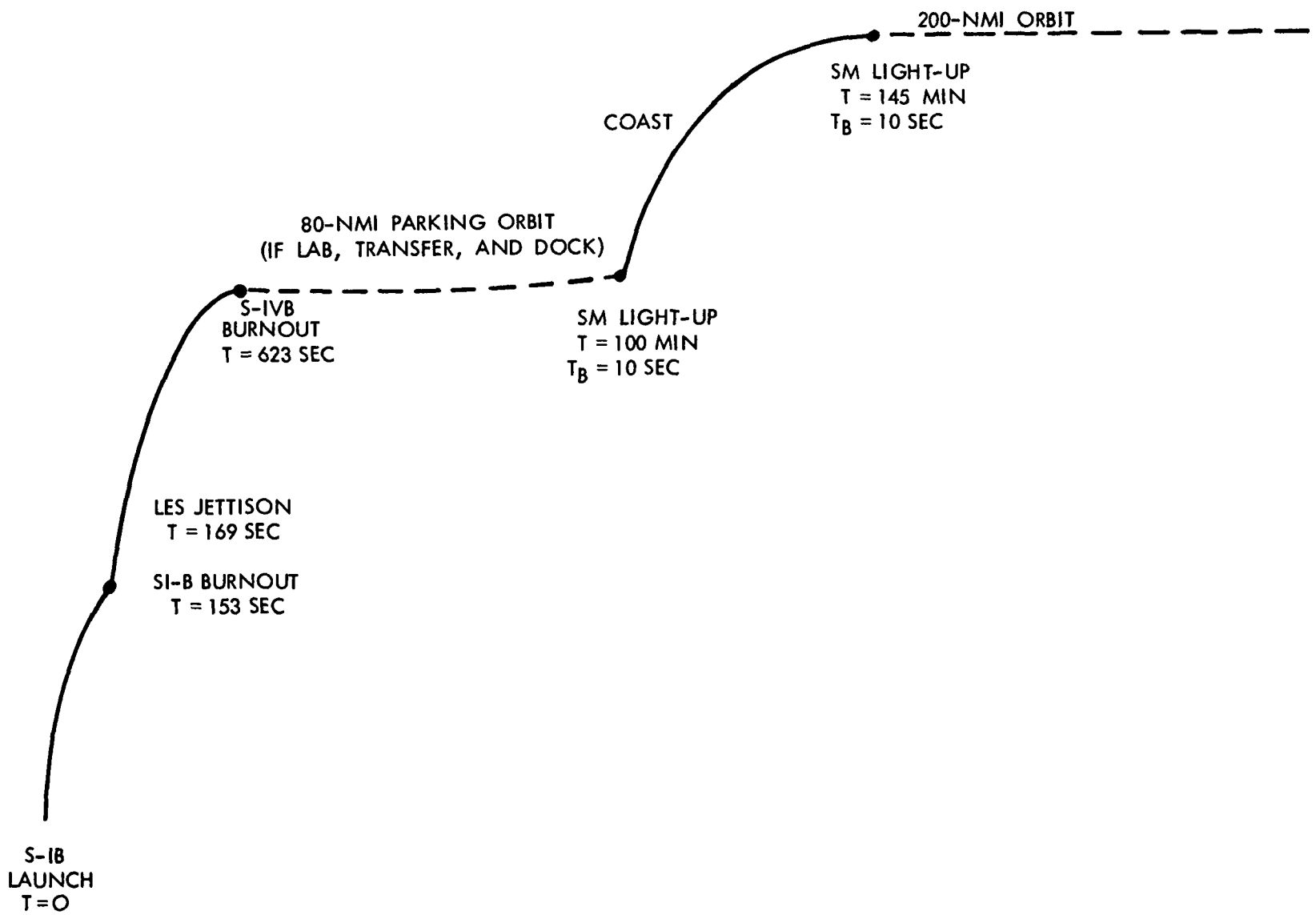

Figure 18. Low Inclination Earth Orbit Mission

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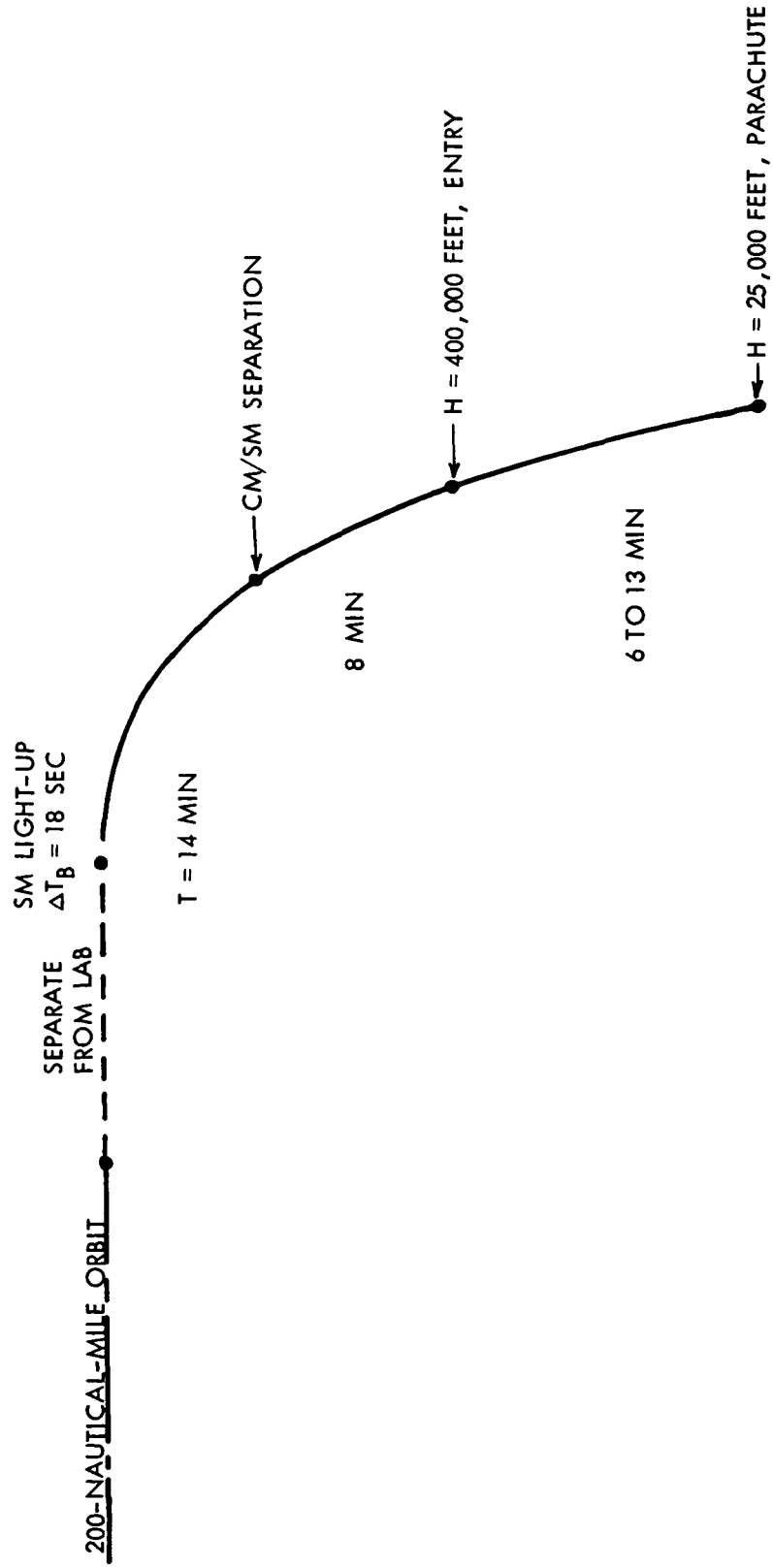

Figure 19. Recovery Flight Profile

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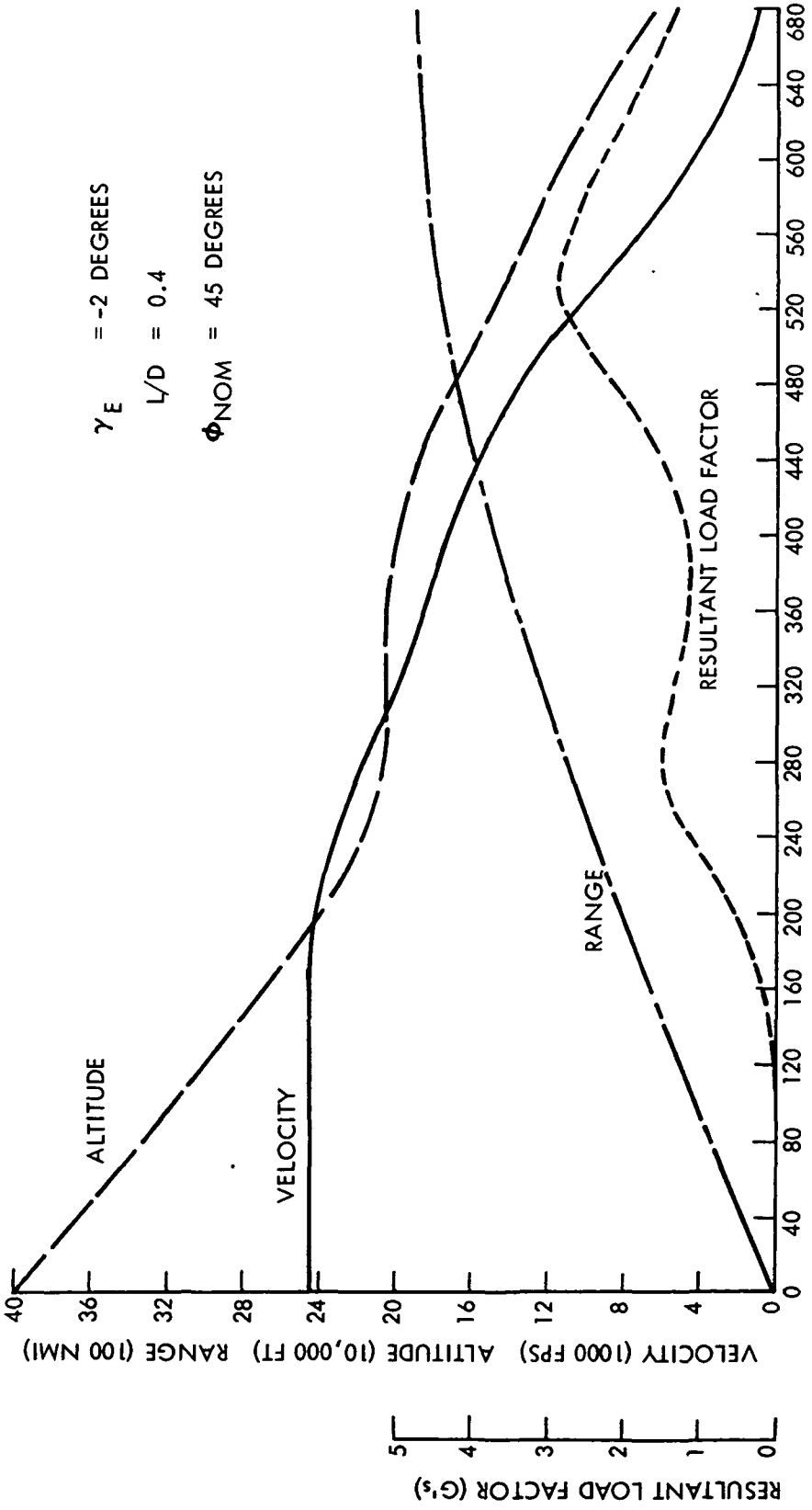
γ_E = -2 DEGREES
L/D = 0.4
Φ_{NOM} = 45 DEGREES
ALTITUDE
VELOCITY
RANGE
RESULTANT LOAD FACTOR
TIME (SECONDS)
0 40 80 120 160 200 240 280 320 360 400 440 480 520 560 600 640 680
VELOCITY (1000 FPS) ALTITUDE (10,000 FT) RANGE (100 NMI)
0 4 8 12 16 20 24 28 32 36 40
RESULTANT LOAD FACTOR (G's)
0 1 2 3 4 5

Figure 20. Nominal Orbit Entry

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POLAR INCLINATIONS

As shown previously in the section on launch vehicle performance, the projected payload weight of the Apollo X is such that the Saturn-V launch vehicle must be used if a polar inclination is desired and if any of the current AMR safety limits are to be observed A typical ascent trajectory and sequence also was presented for the initial orbit attainment. In addition to these analyses, an investigation was conducted to determine the effect of utilizing the Apollo X SPS to increase the available payload at the high-inclination orbits The velocity capabilities of the storable SPS for various configuration weights were determined as shown in Figure 21. When these velocities are expressed in terms of plane change capability with various altitude changes (Figure 22), and used in combination with the Saturn V performance data shown previously for various initial orbit inclinations, the payload weight available for laboratory module and experiments can be determined. This determination is shown in Figure 23 for launch azimuths of 50.0 and 53.0 degrees. The point on the ordinate at a value of 40,000 pounds of SPS propellant represents the limit of initial launch azimuths for the no-laboratory configuration. A design point has been chosen for illustrative purposes, assuming an initial launch azimuth allowable of 53.0 degrees, a laboratory module weight of about 4700 pounds, and a service module configuration with one-half size SPS propellant tankage. The ascent profile for this design is presented in Figure 24. After the first S-IVB stage cutoff, when the orbiting vehicle has established the precision of its orbit by ground tracking and navigational techniques and reached the vicinity of the equator so that the angular change in the orbit will provide the maximum effect on inclination, the S-IVB will be re-started and run to burn-out. At this point the CSM will transpose and dock with the laboratory module, and separate from the spent S-IVB stage and LEM adapter. The service module propulsion will provide the remaining inclination change required at the next equatorial crossing. A geographical representation of the path resulting from this operating sequence is presented in Figure 25 This figure shows an initial launch azimuth of 53 degrees east of north and the concomitant orbit inclination of about 45 degrees. The inclination following the first orbit change is 79.3 degrees and the final change of 10.7 degrees results in the polar orientation. The intermediate orbits shown at the initial 45 degree inclination 105 nautical mile orbit are provided so that orbit determination and guidance system corrections can be made, and so that each subsequent maneuver will be made in the vicinity of a usable ground tracking site.

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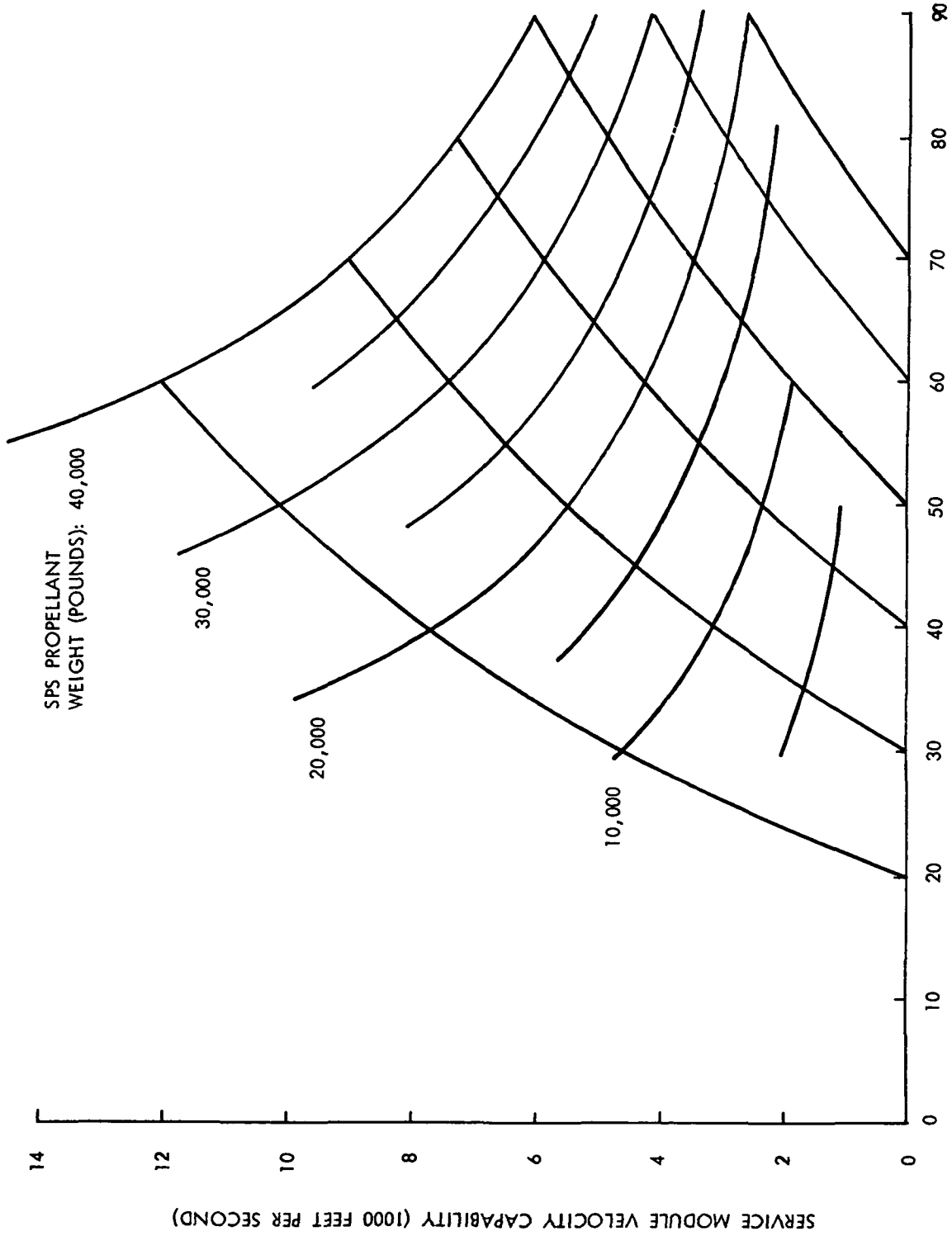

Figure 21. Service Module Velocity Capability

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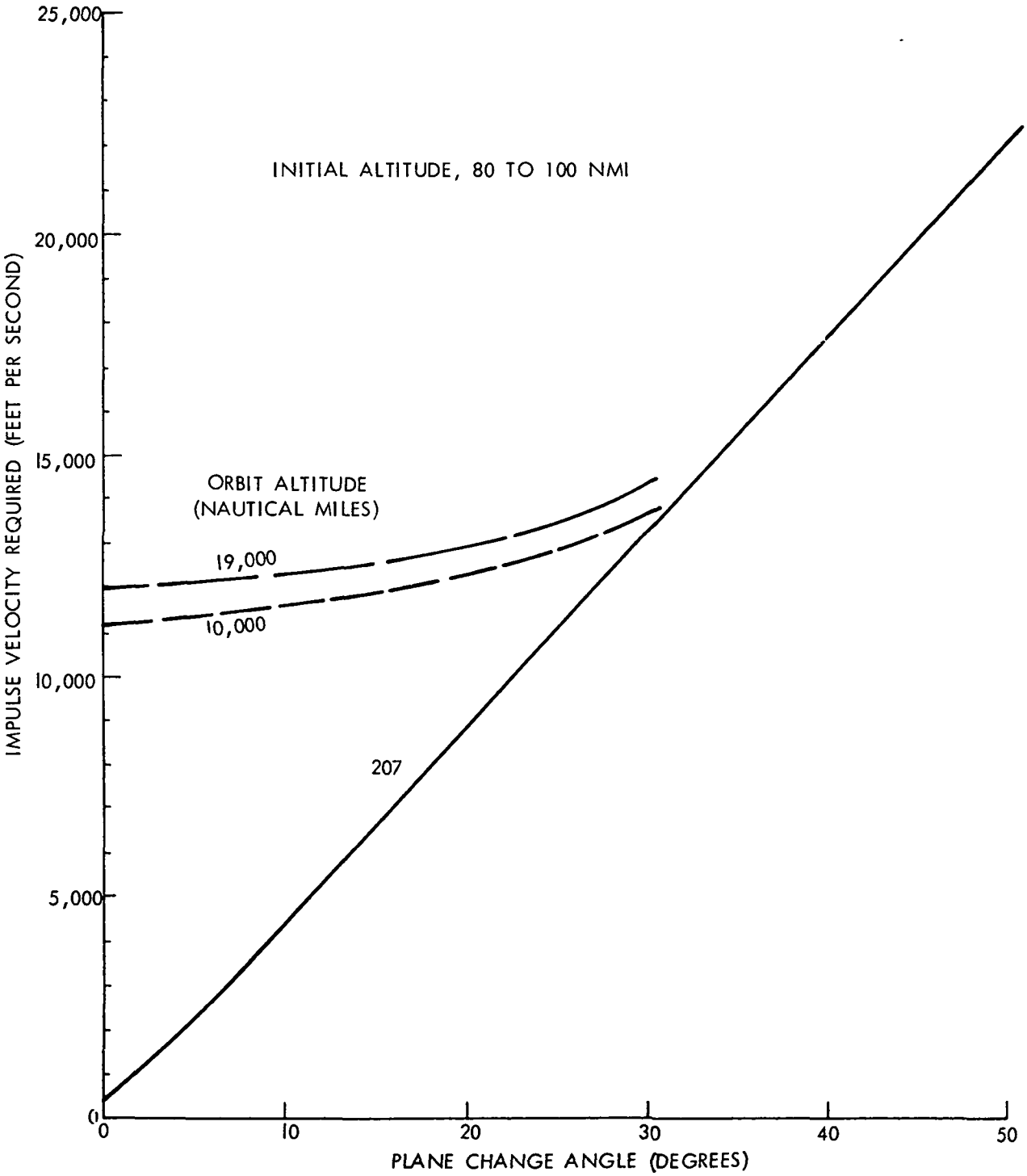

Figure 22. Hohmann Transfer Velocity Requirements

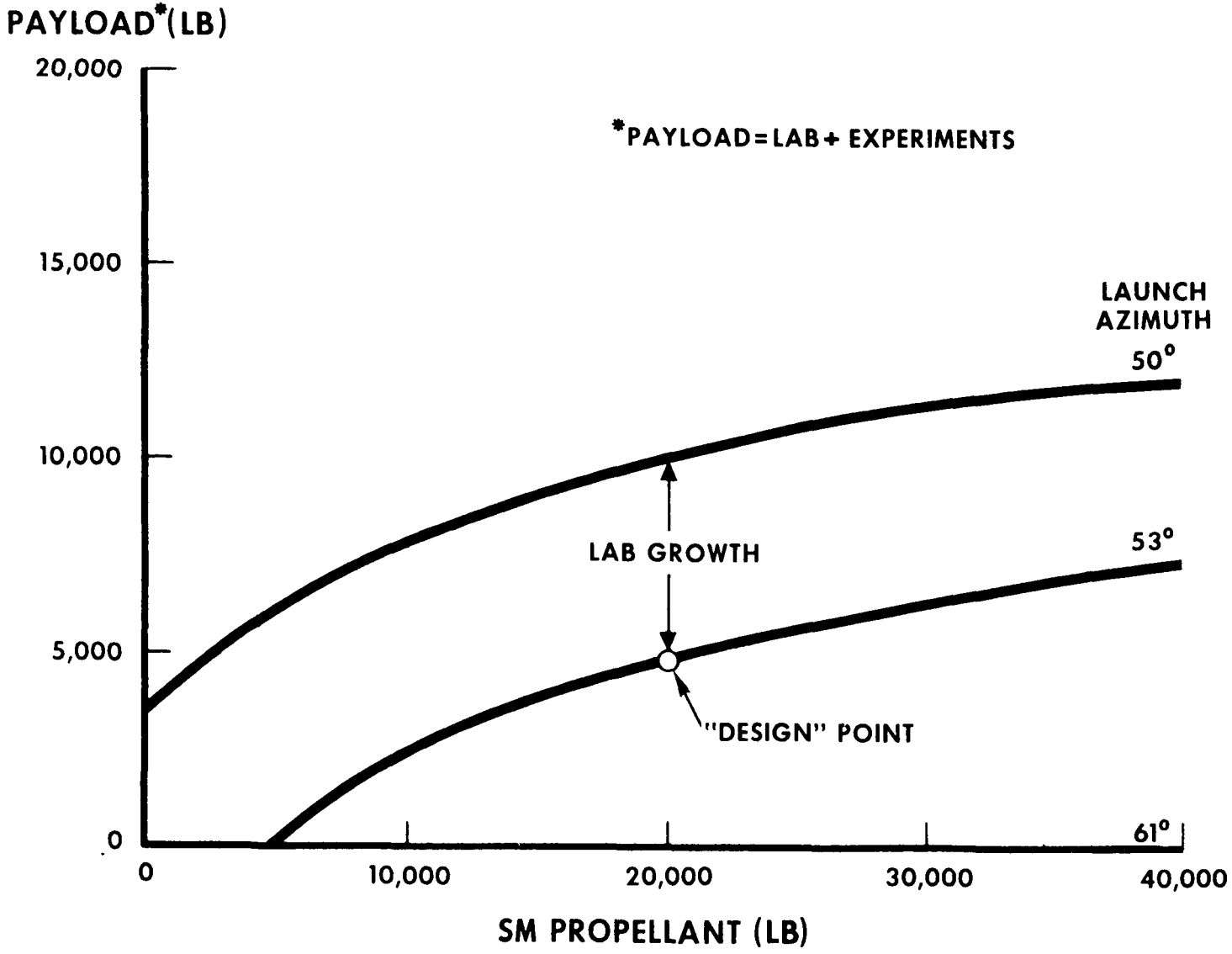

Figure 23. Earth Polar Orbit Capability

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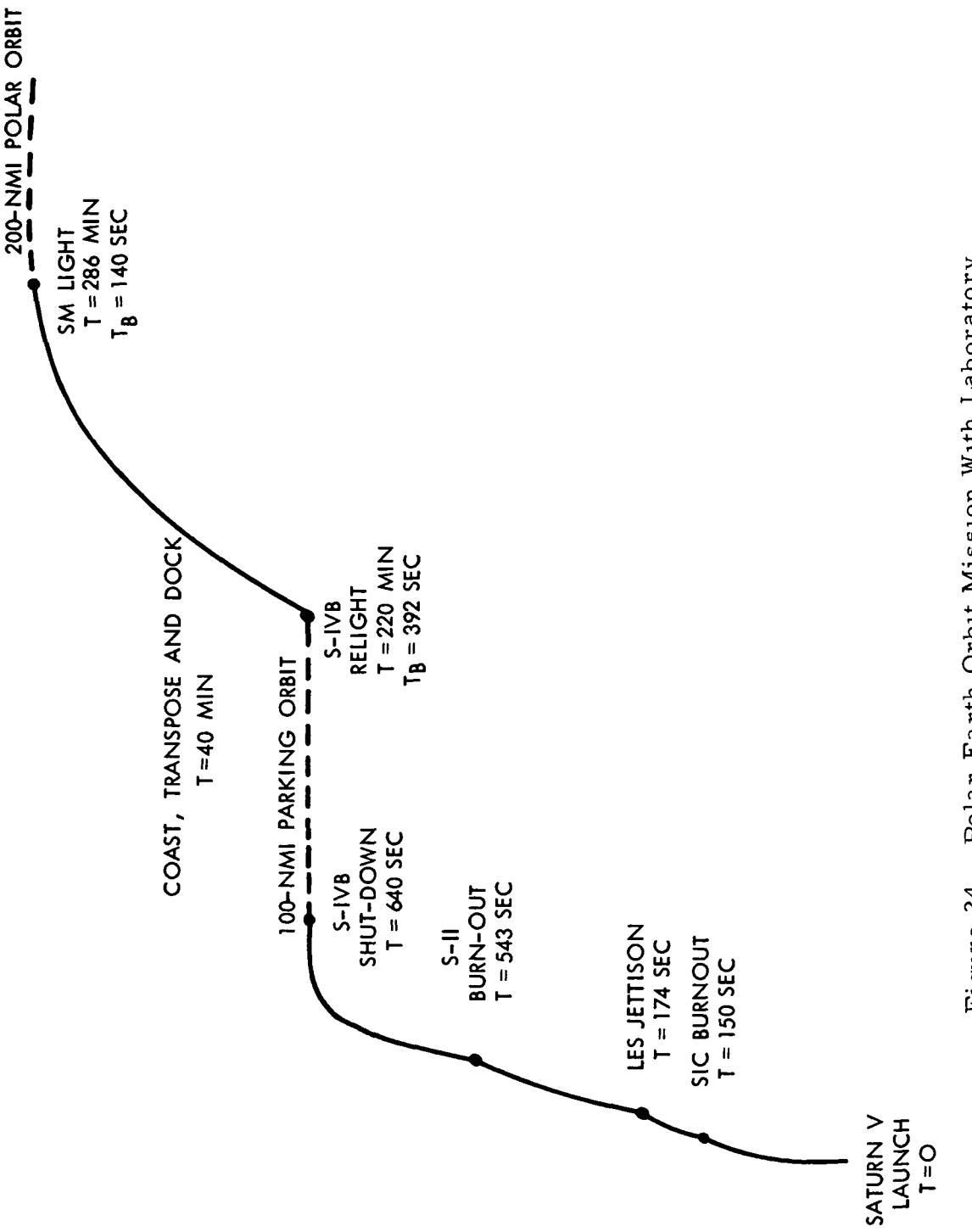

Figure 24. Polar Earth Orbit Mission With Laboratory

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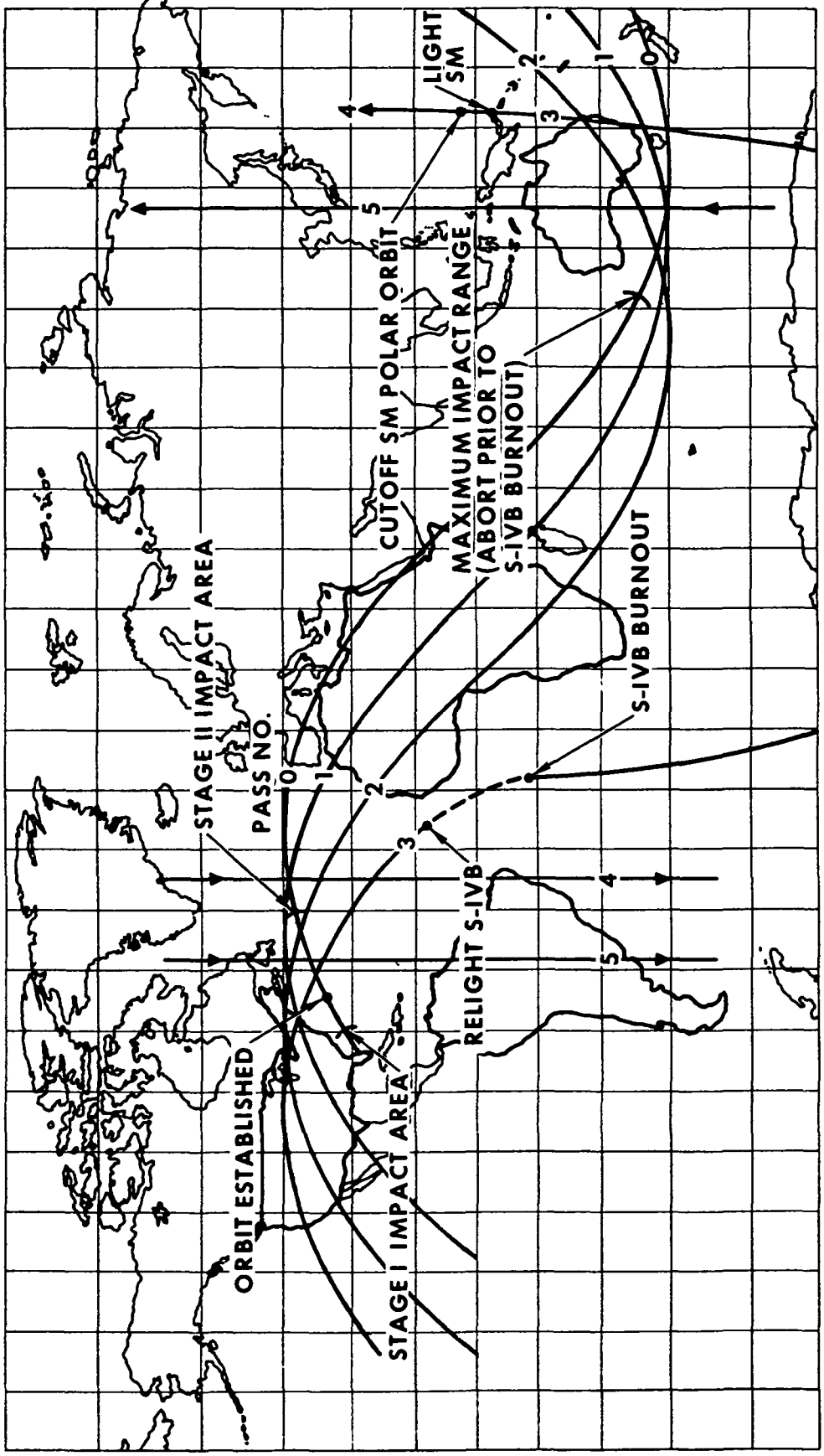

Figure 25 Polar Orbit Ascent

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LUNAR MISSION PERFORMANCE

Within the range of mission time durations postulated for the Apollo X system, the particular mission profile that was found to be critical for the majority of the subsystems was the lunar mapping mission. Because the low inclination lunar orbit performance characteristics are identical to the Block II Apollo, and because mapping of the entire lunar surface is desired, a polar orientation was chosen for examination in this study. Existing performance data developed for the Apollo mission have been utilized, where practical, and any additional information required was developed by using simplified "patched conic" trajectory techniques.

ORBIT LOCATION GEOMETRY

To enable the mapping to be accomplished a particular orientation of the polar lunar orbit is required. Objects being viewed on the lunar surface should have the proper shading so that good sensor resolution will result. The portion of the lunar surface where this shading is found is where the angle of the sun above the horizon is between 15 and 45 degrees. This area is illustrated in Figure 26 by the concentric circles about the sub-solar point. It should be noted that the moon central angle shown is a complement of the aforementioned sun angle.

A typical orbit trace for the initial lunar orbit is also shown in Figure 26. The moon will revolve under the orbit as a function of time since the moon revolves about the earth monthly, and the orbit's orientation is relatively fixed in space. This relative motion will generate the mapping opportunities and will result in each successive orbit trace on the lunar surface having a separation of about 18 nautical miles A second relative motion of the orbit plane, with respect to the lunar subsolar point, will take place as a result of the earth-moon systems yearly motion about the sun. This angular change will amount to about 30 degrees per month and result in the orbit sweeping out a lune of the moon as shown in Figure 26.

LAUNCH WINDOWS

The nodal location of the initial polar lunar orbit discussed above, can be changed by launching the mission at different times or using various trans-lunar transit times. An analysis of the two lunar surface areas discussed indicates that to achieve the maximum overlap, the moon central angle between the initial lunar orbit node and the subsolar point should be 40 degrees west or 70 degrees east If the initial orbit location is held fixed, this

data can be translated into required sun locations with respect to the earth moon line as shown in Figure 27. The velocity vector (V_∞) represents the incoming asymptote of the vertical impact translunar trajectory in relation to the earth-moon line of centers. For the polar lunar orbit under consideration, it also represents the orientation of the initial circular orbit node. The particular angular position of the V_∞ vector is that realized for a 72 hour translunar transit time, and indicates that this mission should be launched so that the sun is either 12, 122, 192, or 302 west of the subearth point at arrival. These locations would correspond to injection at the earth with the sun at about 86, 156, 266, or 336 degrees west of the extended earth-moon line of centers. The variations in the initial orbit orientation due to changes in the earth-moon geometry are 0.536 degrees per hour. Other sensitivities are the same as the present Apollo mission; thus since no definitive criteria exists as to the actual lunar coverage required, the shift produced by these effects is considered negligible.

PAYLOAD CAPABILITY

One significant factor in the launch phase is the booster energy-translunar transit time variation (Figure 28), These data show that a considerable decrease in energy is required for the longer translunar transit times indicating that any additional payload capability desired might be obtained by lengthening this phase of the mission slightly. Other effects such as lunar declination and parking orbit requirements may negate this advantage. Because the Apollo mission is used as a base point where practical, the 72 hour time has been retained for the present study. Approximately the same range of translunar paths is assumed and the polar lunar orbit inclination is provided during the midcourse corrections. The velocity budget for the translunar phase of the mission has been retained from the Apollo at 300 feet-per-second midcourse correction and the additional requirements for the polar orbit estimated at about 80 FPS. The velocity requirements for retro into lunar orbit are shown for an 80 nautical mile perilune altitude in Figure 29. These data presented as a function of translunar transit time are valid for a mean lunar distance. At the base point value of 72 hours, this graph shows a requirement of about 2920 feet-per-second. Summing these velocities with a correction of about 50 feet-per-second for the moon at apogee effect, a total translunar velocity budget of 3350 feet-per-second is defined. To provide some allowance in this figure, the mass ratios determined from this velocity were obtained with a service propulsion system specific impulse of 305 seconds resulting in about a 5 percent margin.

LUNAR ORBIT

The stability of spacecraft orbits has been the subject of past NASA and industry studies. The effects that have been found of significant interest to the performance of satellites with an orbit lifetime such as that of

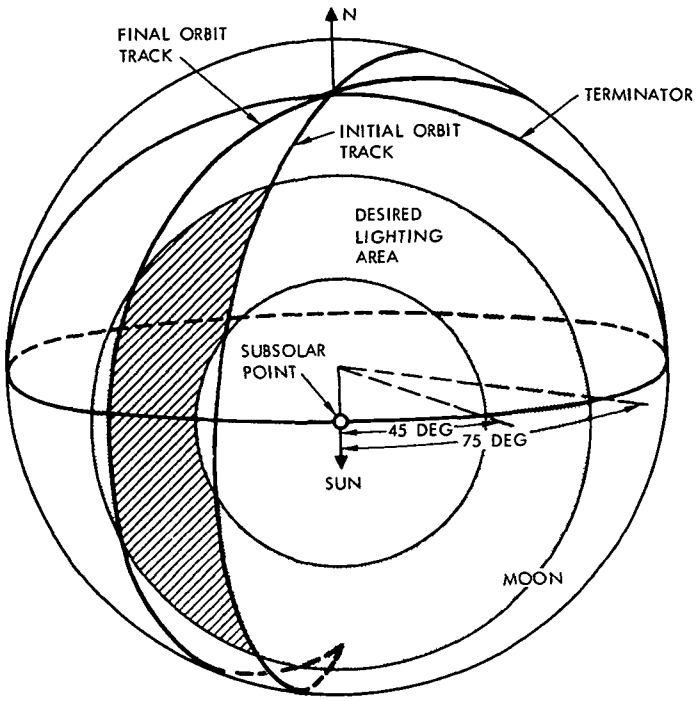

Figure 26. Lighting Characteristics

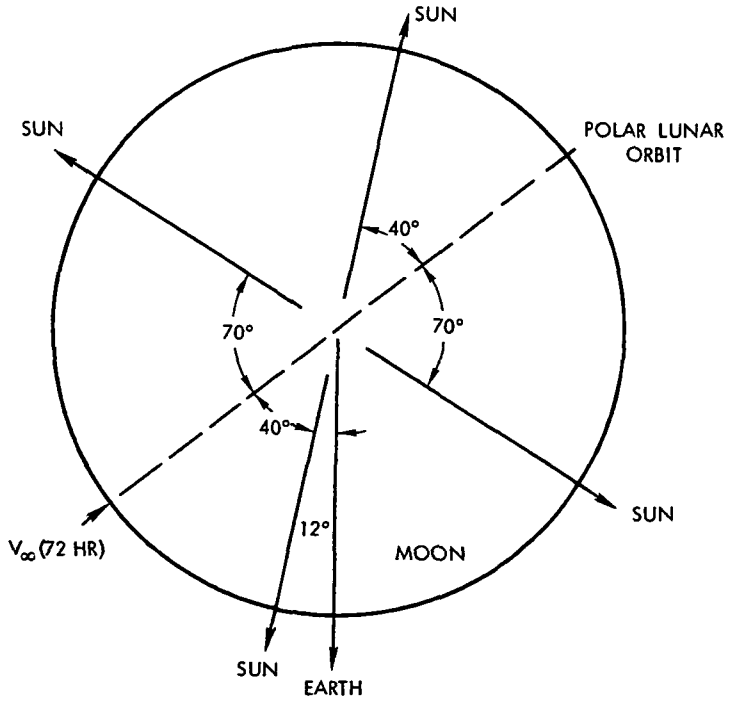

Figure 27. Desired Sun Locations at Lunar Orbit Insertions

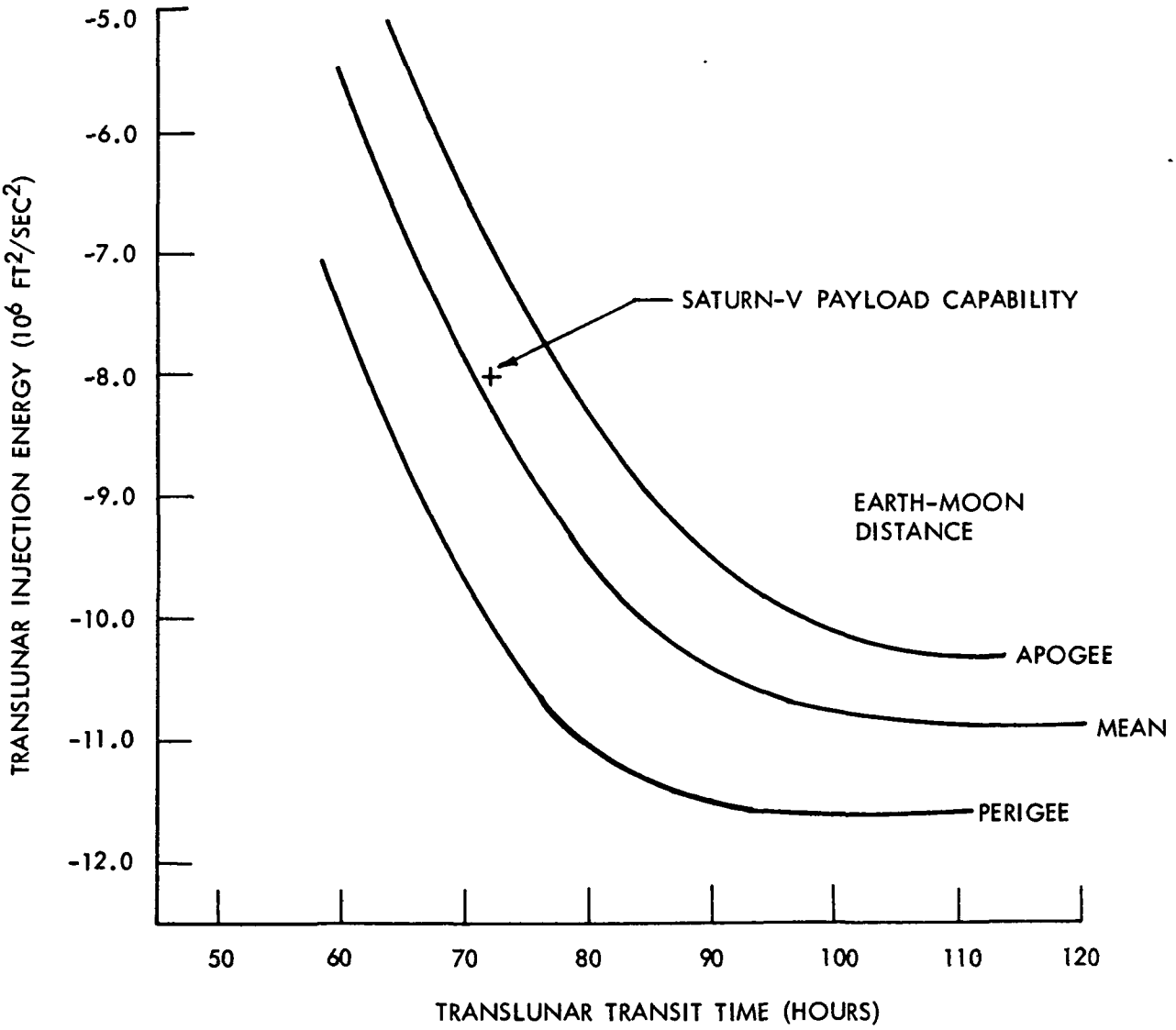

Figure 28. Translunar Injection Energy Requirements

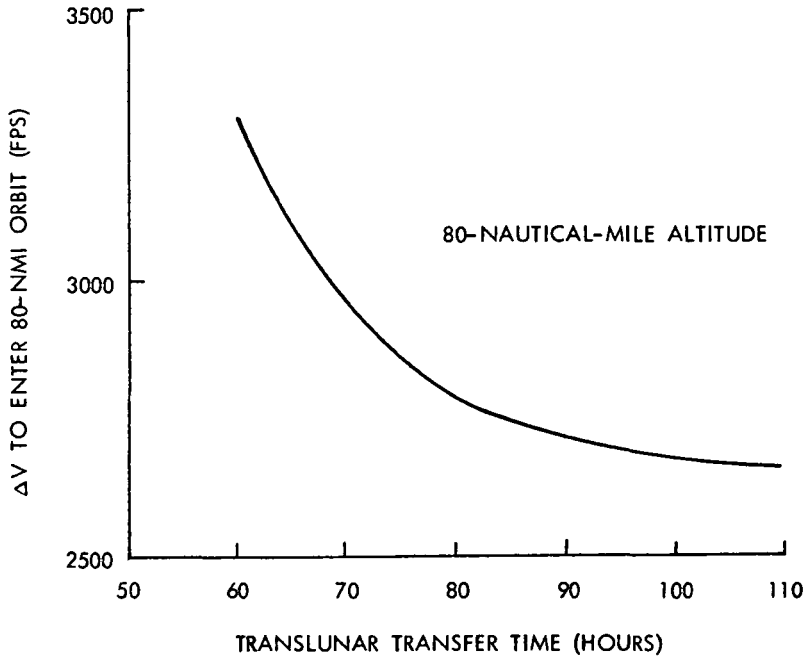

Figure 29. Lunar Retro Velocity Requirements

Apollo X, are those due to aerodynamic drag and planet ablateness. For lunar orbits the first factor is not realized, because the moon has no appreciable atmosphere, thus, only those effects due to the non-homogeneous mass distribution of the moon are present. The major changes in the orbital elements caused by these disturbances, are drifts in the inertial location of the orbit node (intersection with the lunar equator), the eccentricity, and periapsis (minimum altitude) position. Orbit inclination variations are also present having a constant amplitude harmonic form. Simplified analyses show that for nearly circular orbits, the only noteworthy effect is the orbit nodal regression, with its angular rate given by the expression:

$$\Omega = -1.125 \cos i \qquad \text{(degrees-per-day)}$$

For the polar orbit postulated for the Apollo X mission, the inclination angle (i) is 90 degrees, making the cosine term in the above expression zero and resulting in no effective regression rate.

A range of lunar orbits were investigated to substantiate the foregoing analysis. These orbits were computed utilizing an existing IBM machine program, which accounts for the perturbations of the Earth and Sun as well as those mentioned in the previous discussion. An example of the output of this program is presented in Figure 30, for a representative polar lunar orbit over a time duration of 30 days. Review of these data supports the supposition that no station-keeping maneuvers will be required above those for vehicle stabilization and control.

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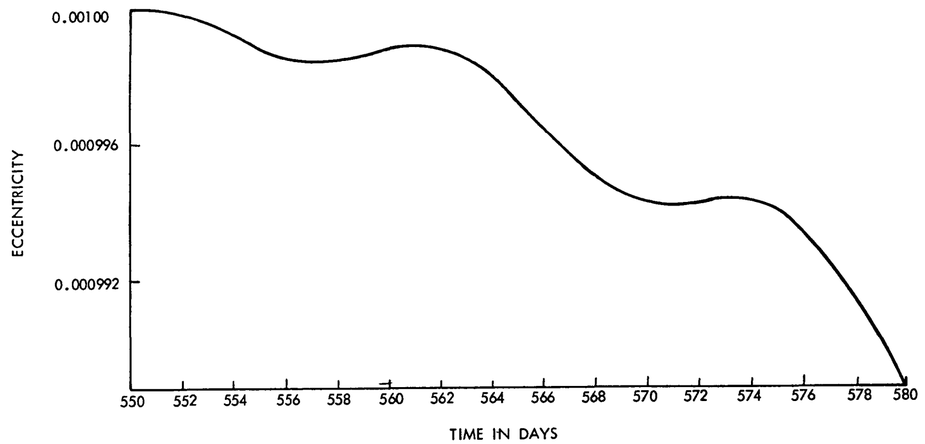

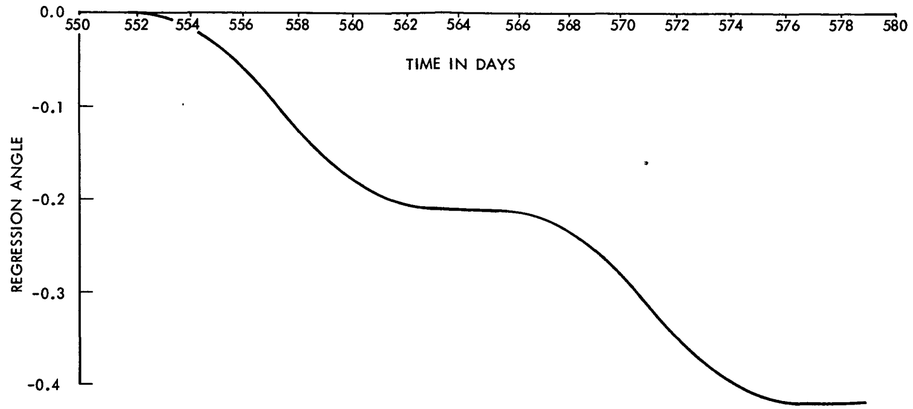

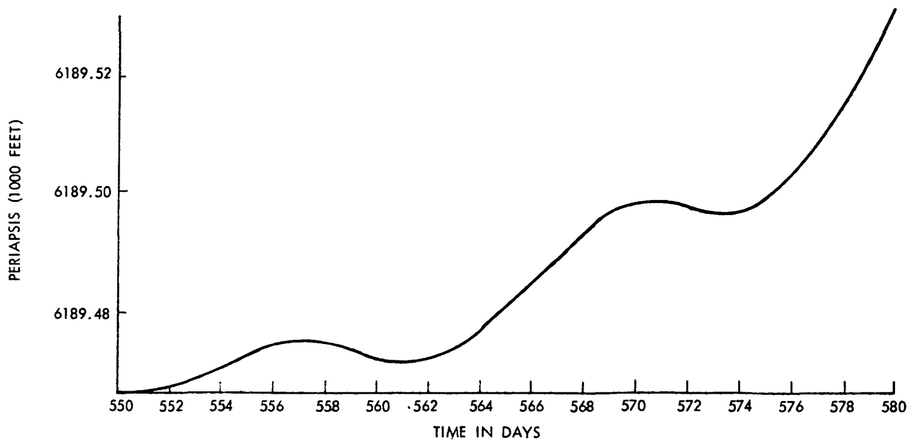

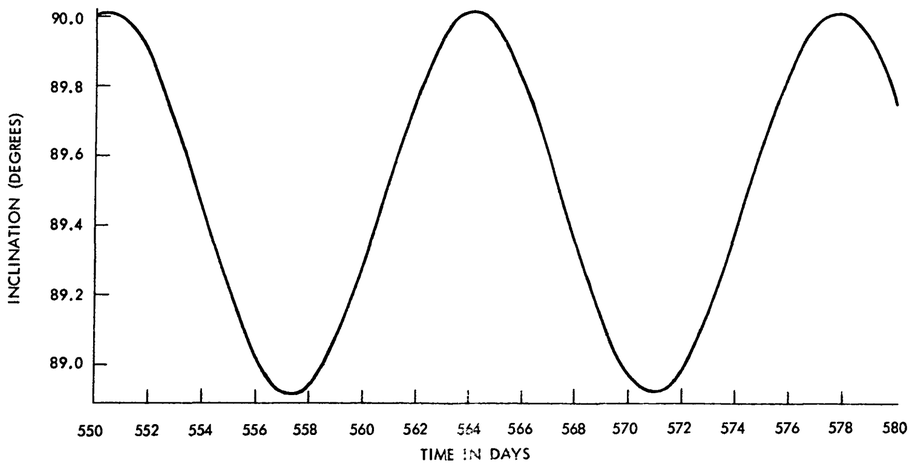

Figure 30. Typical Lunar Orbit History

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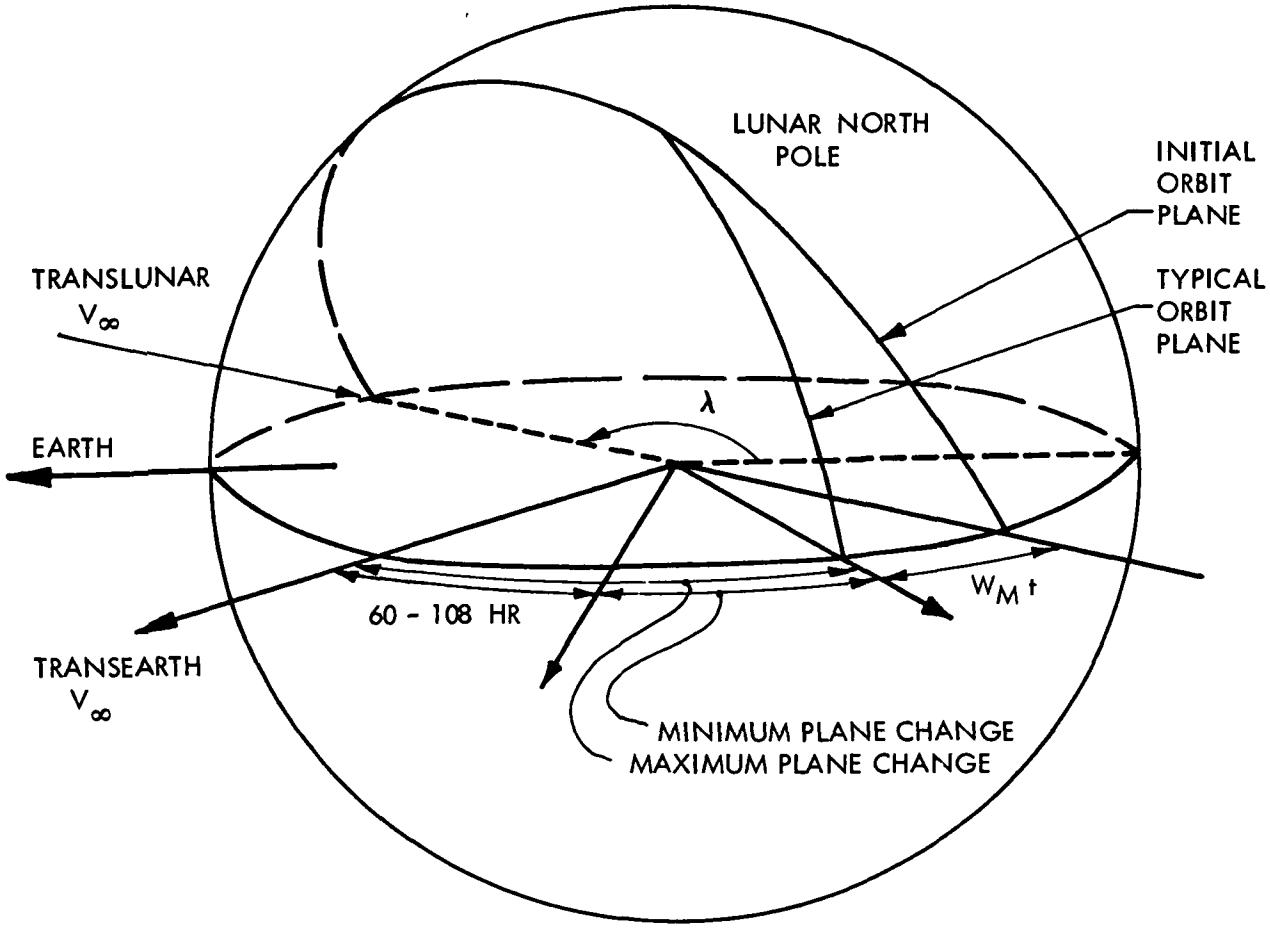
TRANSEARTH PHASE

The trajectory paths that result in acceptable transearth missions have lunar departure asymptote (V_∞) that are in approximately identical orientations with respect to the lunar orbit plane. These directions can be attained from near equatorial lunar orbits (such as that of the basic Apollo), by proper choice of injection time. In polar lunar orbits, which are of interest to the present study, either the lunar orbit node must lie in the required direction or significant changes in the trajectory path must be made.

A geometrical representation of the polar lunar orbit orientation during the lunar stay is shown in Figure 31. The magnitude of the initial nodal angle, λ, (measured east at the moon from the extended earth-moon line of centers), is determined by the translunar trajectory characteristics of injection velocity and transit time. Nodal angles corresponding to the orientations of the V_∞ vector that are required for a range of transearth transit times are also shown. Values of 219 and 263 degrees were determined for respective transit times of 60 and 108 hours.

As the vehicle stays in lunar orbit, the nodal position changes as shown in Figure 32 at a rate equal to the moon's rotation rate relative to the earth. After a stay time in lunar orbit of about 3.5 days the orbit coincides with the required direction for a 108 hour return trajectory and a low-energy return can be initiated. By changing the return transit time from 108 to 60 hours corresponding with the further nodal change, a reasonable velocity requirement for transearth injection can be retained up to about seven days. Within this area of nodal positions just discussed, the return trajectories would be almost identical to those of the basic Apollo. Further motion of the nodal location would require that plane changes be made in addition to the nominal injection requirements to provide the proper return trajectory orientation. A graphical representation of these results is given in Figure 32. These data show the angle between the present nodal location and that desired for return as a function of orbit stay time for the range of transit times considered. These data have a cyclical variation with a period of one-half month and are representative of a mean lunar distance.

Three general methods can be postulated to achieve the desired return orientation and velocity. The first and relatively straight-forward approach would be to simply change the lunar plane to bring it into the desired return direction, and then make the transearth injection maneuver from this new orbit. The velocity required in addition to the basic values of 2920 feet per second for injection, 80 feet per second for the lunar distance effect, and 300 feet per second midcourse correction, is shown in Figure 33 for the range of return times being considered.



W_M = MOON'S ROTATION RATE

t = ORBIT STAY TIME

λ = RELATIVE LONGITUDE OF APPROACH AND DEPARTURE VERTICAL IMPACT PTS

Figure 31. Lunar Orbit Geometry

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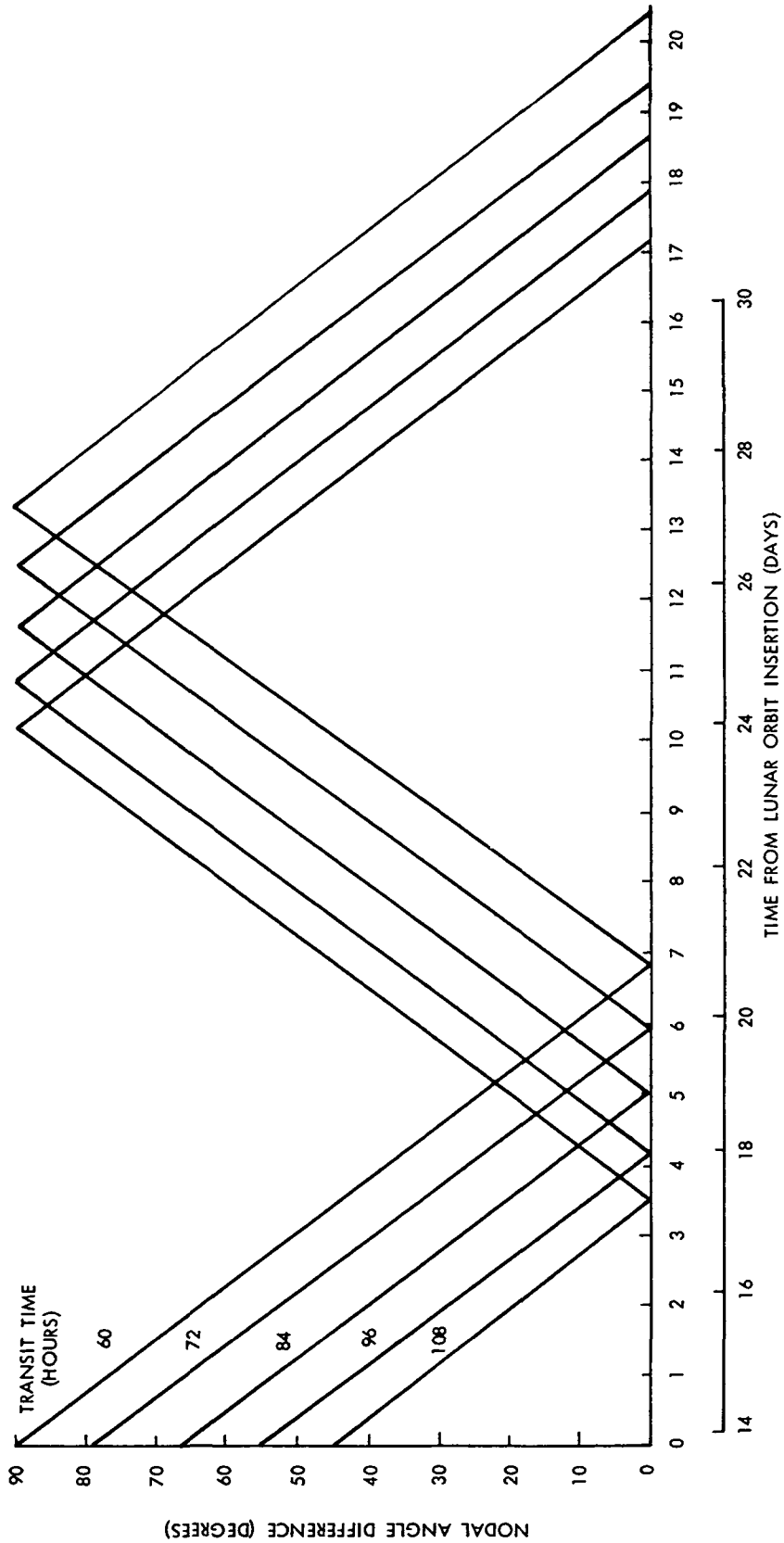

Figure 32. Nodal Angle Characteristics

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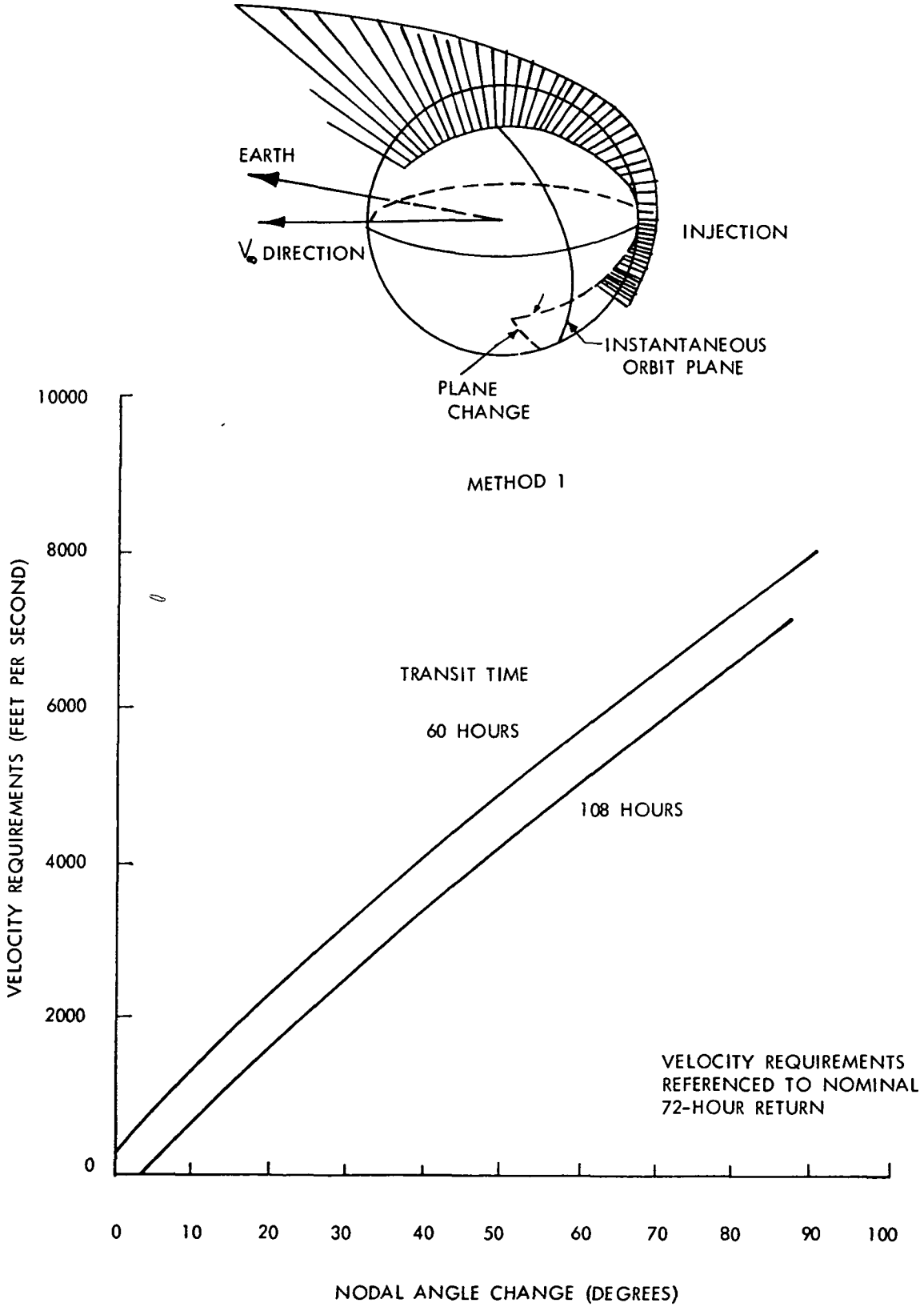

Figure 33. Transearth Excess Energy Requirements, Method I

A second method to achieve the desired V_∞ direction from various orbital positions would be to make the return trajectory orientation change at the same point as the transearth injection so that the vector addition characteristics would decrease the separate velocity requirements. The results obtained from an investigation of this procedure are given in Figure 34. A comparison of these data with those of the first method (Figure 33), indicate that a lower velocity is required at the small angles, but the latter method is definitely better from an energy standpoint for angular changes greater than about 40 or 50 degrees.

The third method investigated and that chosen for use in further analyses, is presented in Figure 35. In this approach, the initial transearth injection is made as if the instantaneous lunar orbit plane were in the correct orientation, and with a second maneuver at the lunar sphere of influence the trajectory is brought into coincidence with the desired return direction. A comparison of the methods postulated shows that the second method requires the least energy for low angles of change; for angles higher than about 20 degrees, where significant energy expenditure is required, the third method is preferable.

When these maneuvering energy characteristics are related to the geometry shown in Figure 32, a time history of return velocity requirements is obtained (Figure 36). The curves shown represent the range of return transit times mentioned previously. The rather large velocity variations are due to the different effective nodal angle changes necessary to establish the required transearth trajectory plane orientation. A review of these data shows that the stay time periods between 3.5 and 7 days, and 17.2 and 20.7 days, require only enough velocity necessary to achieve the return transit time. These trajectory paths would be equivalent to the present Apollo mission. As the stay time in lunar orbit extends beyond these periods the maneuver requirements increase, with a maximum of 9600 feet per second required to utilize the 60-hour return time for the entire month.

LUNAR ABORT CONSIDERATIONS

The velocity and return time characteristics for aborts during the translunar phase of the mission are shown in Figure 37, as a function of time from injection. A nominal Apollo X maneuver capability at this point in the mission would be about 9000 feet per second, which, by the data shown in the figure, would provide return trajectory times of less than 40 hours. The curve data is only carried to about 50 hours from injection, because beyond this point the lunar influence invalidates the simplified analysis technique, and although direct aborts can be initiated from well into the lunar vicinity the return time-velocity characteristics will be changed from those computed by this technique.

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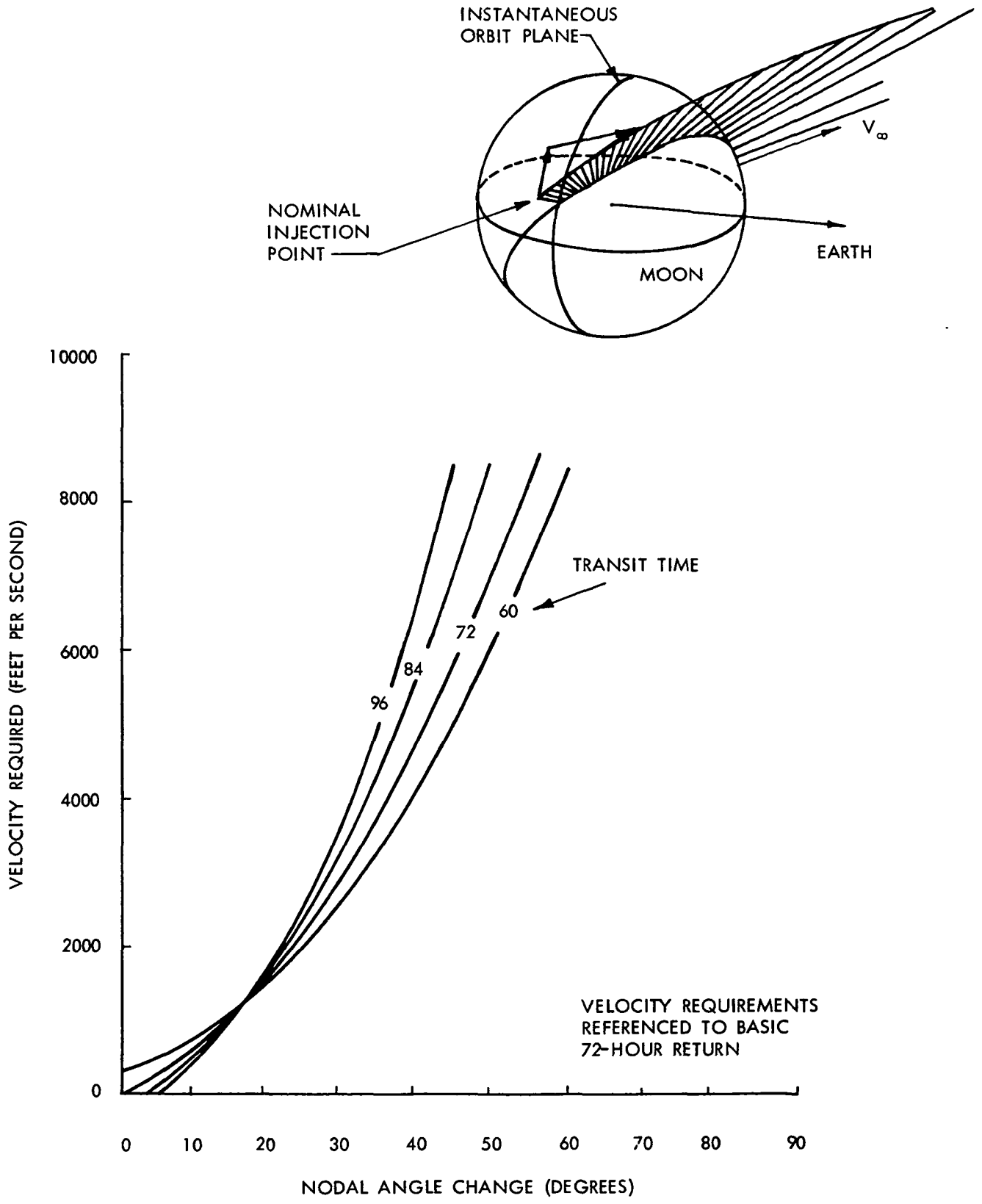

Figure 34. Transearth Excess Energy Requirements, Method 2

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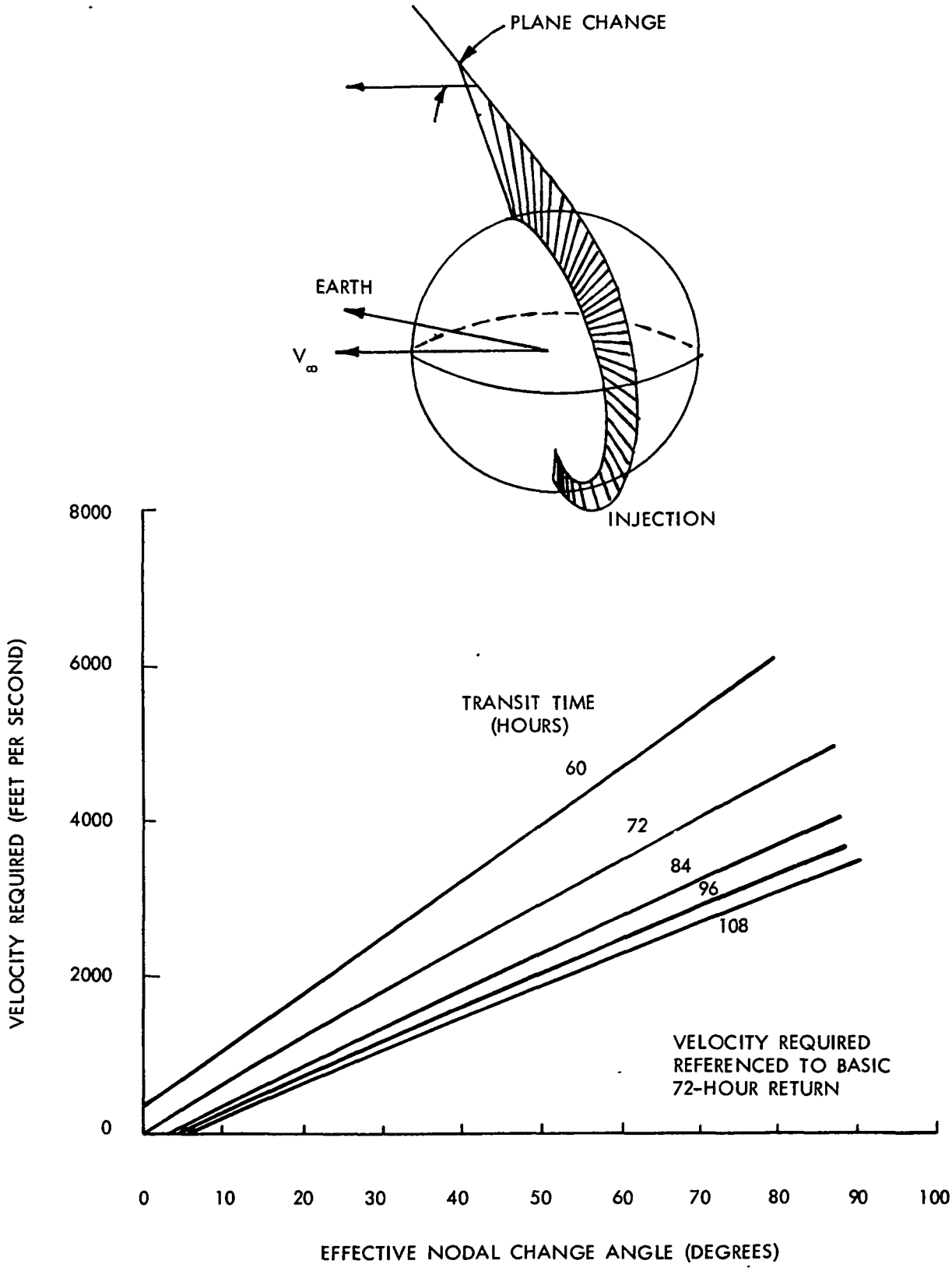

Figure 35. Transearth Excess Energy Requirements, Method 3

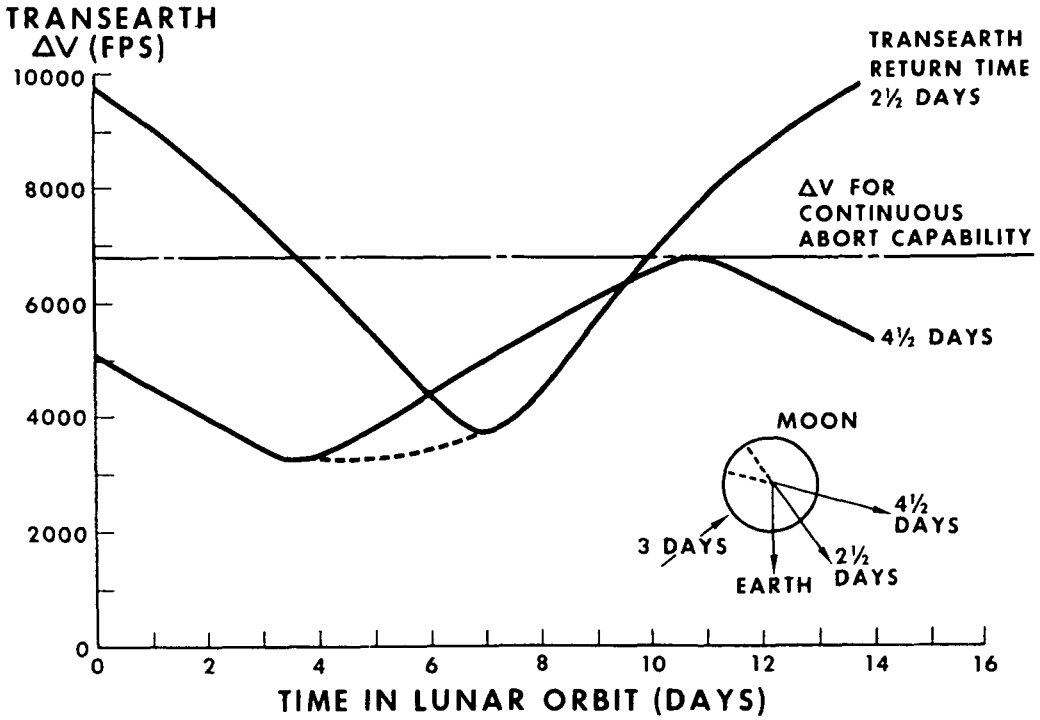

Figure 36. Abort Requirements, Lunar Polar Orbit

Once the lunar orbit has been established, the abort energy requirements are equivalent to those shown in Figure 36 for the normal return phase with the exception that the entire range of lunar orbit stay times must be considered. Review of these data show that a minimum velocity capability of 6600 feet-per-second must be provided to allow a return at any time of the month. This value also requires the utilization of the return time flexibility of 60 to 108 hours. The monthly variation of this abort time, for an assumed 6600 feet-per-second return capability is given in Figure 38. These data indicate that only small time spans will require the use of abort transit times above 3 or 3-1/2 days with a large portion of the stay time spectrum requiring less than these values.

The projected Apollo X weight variations were used to estimate the useful payload (laboratory module plus experiments) available as a function of the return time characteristics (Figure 39). The payload shown is assumed to be left in lunar orbit. Because the basic vehicle weight decreases as a function of time due to the consumption of expendables, the two dips in the curve reflect the sensitivity of payload to this variable. The dip to zero indicates that for all payload values some time period will exist during which a return cannot be initiated. For the Apollo X useful payload weight of 4700 pounds that has

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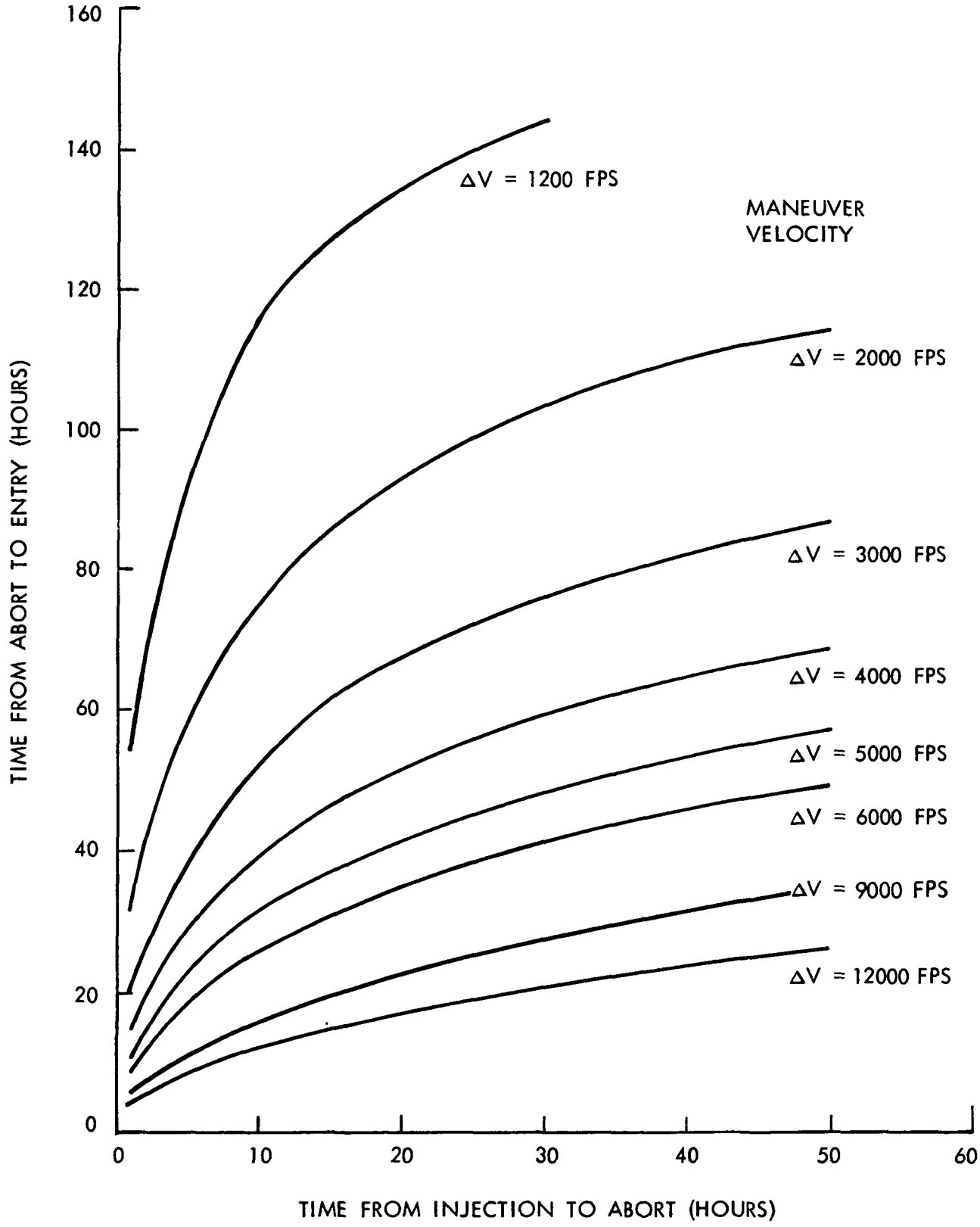

Figure 37. Translunar Direct Orbit ΔV Requirements

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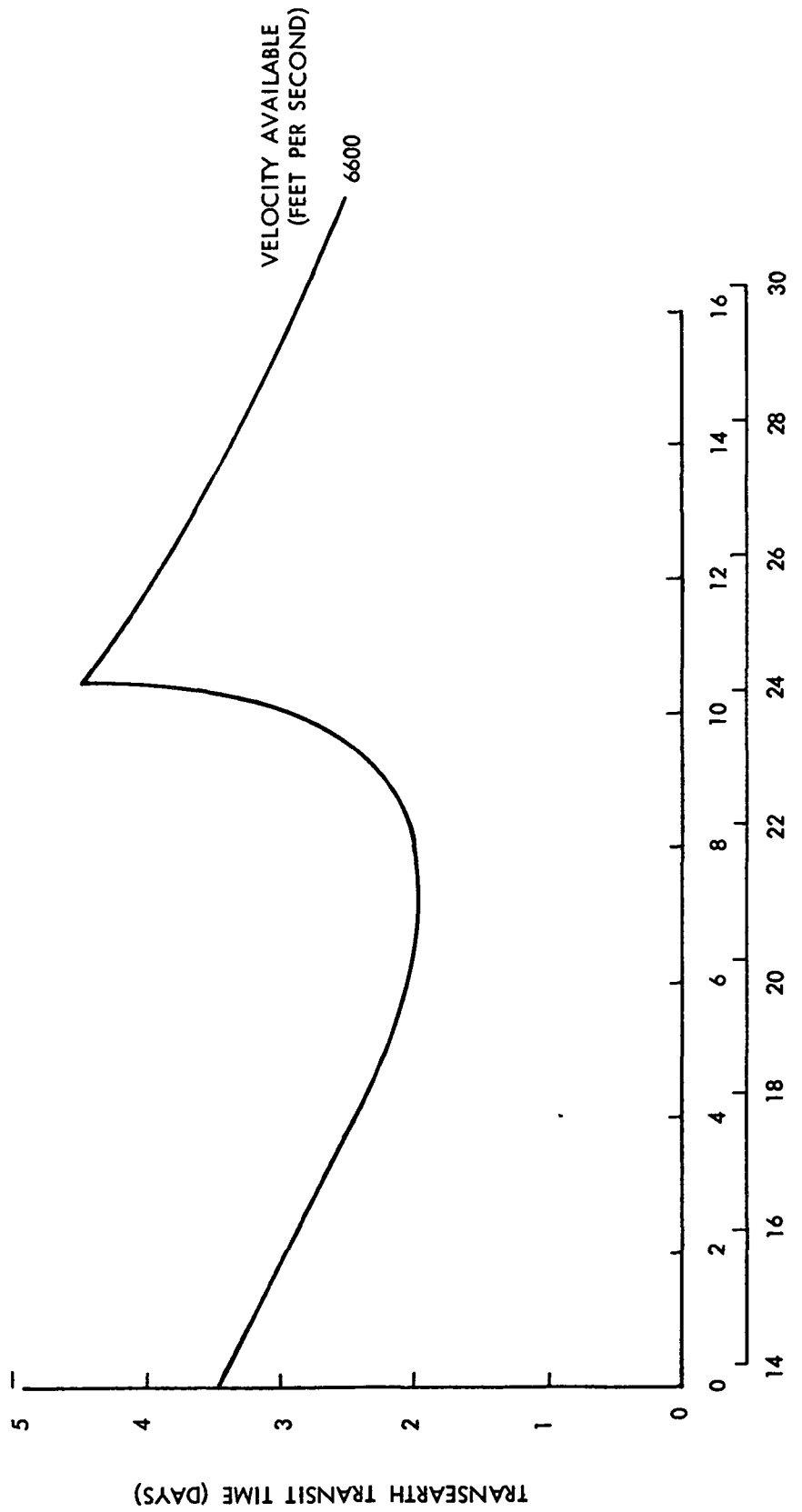

Figure 38. Abort Time Characteristics

been estimated by present study efforts, a time period of about two days (from 9.5 to 11.5 days after arrival in lunar orbit) exists during which a return will not be possible. If return initiation can be restricted to short time periods, the data show that payloads of above 40,000 pounds could be accommodated.

ABORT WITH ALTERNATE PROPULSION CONCEPTS

The configurations whose performance was discussed above were limited in payload or maneuver capability by the Service Module propellant capacity and did not approach the limits of boost payload; therefore, the use of additional propulsion was investigated. The LEM descent stage or solid motors were considered for use during either lunar orbit insertion or escape from lunar orbit. The results of using the LEM Descent stage in these two modes are also shown in Figure 39. Because the first mode of operation requires a two-burn retro into lunar orbit, separated by a staging sequence and transposition maneuver, the magnitude of the first burn was chosen to provide at least a low eccentricity lunar orbit such that if trouble was encountered in the staging or transposition, a number of waiting orbits could be utilized before making the second maneuver to establish the circular orbit. This intermediate orbit would have an apo apsis below 3000 nautical miles. The data of Figure 39 shows a payload capability of about 12, 000 pounds can be achieved while maintaining the continuous abort capability.

Another mode of operation considered was that of using the LEM Descent stage for the initial transearth maneuver. The characteristics of this method are also presented in Figure 39 and indicate an increased capability of only about one-third as much as the previously discussed approach. This result is caused by the fact that the LEM stage is mounted on the Laboratory Module and must also boost this payload out of the lunar orbit.

The use of solid motors in place of the LEM Descent stage just discussed, provides about the same results, because, although the specific impulse of these units is lower, the mass ratios are somewhat higher. Because of this performance similarity and the fact that the solid motors are not man-rated, they are not presently being considered for the Apollo X.

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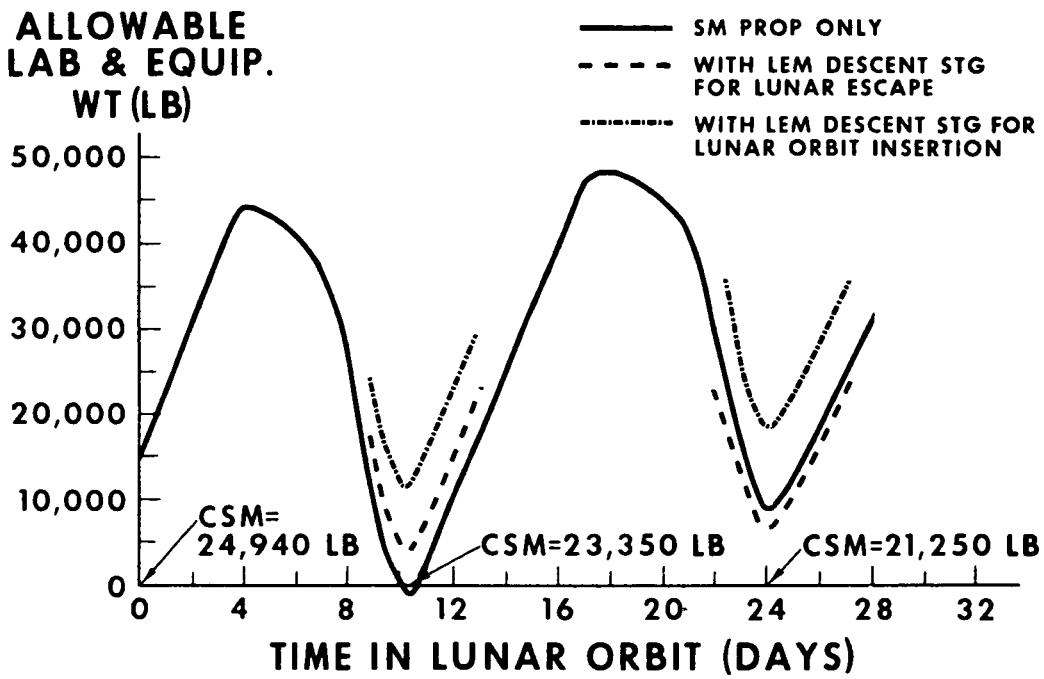

Figure 39. Lunar Polar Orbit Payload

COST EFFECTIVENESS ANALYSIS

The cost effectiveness analysis described in this section is intended to clarify the cost variations that can result from two different approaches to the design of an Apollo X earth-orbital experimental laboratory. The first approach is to use the configuration of the 28-day lunar mapping vehicle as the design goal. The cost of a 45-day earth-orbiting vehicle is then found by costing the changes to the Block II Apollo that enable the vehicle to perform a 45-day earth mission and that are consistent with a minimal set of modifications culminating in a lunar mapping configuration. Vehicles both with and without a laboratory module are considered.

A slightly different alternative is to cost the modifications that enable the Block II Apollo to perform an earth mission of between 34 days and 45 days duration both with and without a lab module. As before, the modifications are part of a minimal set culminating in a lunar mapping configuration.

The second major approach is to use the earth-orbiting Apollo X as the design goal. The cost of this vehicle is found by costing the modifications that enable the Block II Apollo to perform a 45-day earth mission both with and without a laboratory module with no consideration given to eventual lunar mapping capability.

There are, therefore, six vehicle configurations for which costs are made available for comparisons. Comparisons are made between configurations to show the cost increments that result from adding a laboratory module. Cost difference, if any, are also shown between the limited orbit duration vehicle oriented toward eventual lunar mapping and the corresponding 45-day vehicle. Finally, the cost effects of eliminating lunar mapping considerations entirely are evaluated.

In all cases, specific mission demands on the vehicle systems are suppressed by assuming that the missions are limited to the performance of a biomedical and behavioral experiment package appropriate to the available volume and conducted in a 200-nautical-mile, 28.5-degree orbit. This expedient has the advantage of removing cost variations resulting from special mission requirements and, hence, of simplifying the comparisons between the two major approaches previously defined. It can also be justified on the basis that the biomedical and behavior experiments are the only ones for

which a quantitative measure of effectiveness has been developed. Nevertheless, there is a serious pitfall in that conclusions drawn from such an analysis may not be valid for situations involving other types of missions and particularly those missions dependent upon orbit inclination, maneuvers, available electrical power, etc.

MEASURES OF EFFECTIVENESS

In addition to obvious measures of performance effectiveness for comparing the six defined configurations, (i.e., allowable net payload weight and allowable payload volume for experiments) the product value of the biomedical and human factors experiments is assessed by utilizing some of the results of the Biomedical and Human Factors Requirements Study (SID 63-1392) performed by S&ID under NASA contract NASw-775. As far as can be determined, there is no difference in the BAHFR experiments that can be performed in any of the 3 configurations involving a laboratory module and, likewise, for the three involving only the command and service module. Consequently, the experiment measure-of-effectiveness values will be useful only in comparing performance differences between laboratory and no-laboratory configurations. The allowable payload weights and volumes must be used for comparing the three design approaches

MEASURE OF EFFECTIVENESS FOR BIOMEDICAL EXPERIMENTS

In the BAHFR study, a list of 97 separate biomedical measures (tests) were compiled. This list was assumed to include all biomedical measures that had any bearing whatsoever in determining long-time zero-g effects on the crew To each separate measure, a weight was assigned by a complicated procedure, which will not be described here The sum of the weights was exactly 1.000 with a weighting distribution nearly logarithmic (i.e., the first measure value having a value approximately 100 times greater than the last measure).

During the present effort, the list has been slightly refined with several measures added and some removed The end result is a list of 105 measures approximately logarithmically weighted with the weight sum equal to the 1.000 mentioned before There are a few measures that cannot be performed in either configuration. Most of these measures are eliminated, because they require an attending physician. The complete revised list is given in Volume 3 of this report. Because the relative performance between the laboratory and no-laboratory cases is of interest here, only the measures and their weights that cannot be performed in either configuration or that can be performed only with a laboratory are listed below. The reasoning involved in the selection is described fully in Volume 3 of this report.

Measures that cannot be performed in either configuration of Apollo X	Weight of Measure
Cardiac output	0.0982
Total blood volume	0.0179
Exercise tests	0.0162
Venous pressure and circulation time	0.0142
Tubular excretion	0.0085
Catecholamines, serum	0.0050
Catecholamines, urine	0.0040
17-ketosteroids, serum	0.0040
17-ketosteroids, urine	0.0033
Liver function	0.0025
Mineral corticoids	0.0018
Pulse wave velocity	0.0012
Total weight of excluded measures	0.1768

Additional measures that can be performed only in laboratory module configurations	Weight of Measure
Centrifuge test	0.0315
Venous pCO_2, pO_2, and pH	0.0200
Tubular reabsorption	0.0085
Serum asmolarity	0.0050
Oxygen uptake, RBC	0.0050
Biocarbonates, serum, saliva	0.0028
Transaminase, blood, saliva	0.0026
Capillary morphology	0.0012
Heart sounds	0.0071
Heart force	0.0024
Red blood cell mass	0.0055
Iodine uptake, RBC	0.0043
Red blood cell survival	0.0041
Pulmonary pathology	0.0162
Bone density	0.0120
Kidney stone formation	0.0056
Total weight of measures limited to laboratory configurations	0.1338

Therefore, 1.0000
0.1768

0.8232 = Total weight of measures in Apollo X laboratory configurations

0.8232
0.1338

0.6894 = Total weight of measures in Apollo X command and service module configurations

Relative effectiveness of command service module configuration as compared to laboratory configuration $\frac{0.6894}{0.8232} = 0.8375$

MEASURE OF EFFECTIVENESS FOR BEHAVIORAL EXPERIMENTS

In order to explain the development of a measure of effectiveness for the behavior tests, it is necessary to briefly explain the procedure that was followed in the BAHFR study in assigning so-called confidence weights to the tests

Fifty behavioral functions were chosen, an investigation of which was believed to provide the answers to all possible questions concerning degradation in behavior caused by weightlessness Each function was equally weighted (i.e., assigned a weight of 0 02). To each function there was also assigned a measure (test) that was believed to be the best or primary method of evaluating the degrading effect of weightlessness on that particular function. Most of the measures had some value in evaluating the degradation of one or more of other functions in addition to the primary one associated with it For each such measure-function relationship, probability values p and q were assigned The validity number p indicated the probability that the measure would indeed detect a change in the function caused by weightlessness The quality number q was an independent estimation of the ability of the measure to provide a true value of the detected change

The method proceeded as follows All functions that could be evaluated by a given measure were listed under the measure together with the product p x q x 0 02 = W as calculated for each function relevant to that measure The sum of the W's was defined as the contribution of that particular measure to the total confidence associated with a set of measures if none of the functions were measured by more than one test Because this restriction usually did not apply, a more general definition of the confidence of a given set of measures was adopted (i.e., the confidence level of a set of measures was defined as the sum of the highest W values for each of the functions listed under any of the measures included in the set) In this way the total confidence value of the set of 50 tests was found to be 0 80

Finally, the tests were ranked in order of importance under the assumption of a time limitation That is, if the available testing time was less than the time required to perform all the measurements, the subset

selected should be chosen by working down the list from rank one until all the available time has been expended.

During the present study, several measurements were deleted and some were added. The result was a list of 52 functions that are to be measured for a complete investigation of the behavioral effects of weightlessness Corresponding to each of these 52 functions is a measure (test) that best evaluates that function Most of the tests are useful in checking more than one function. For each measure-function relationship, a k value is assigned that replaces the pq product previously used but that retains the sense of both measure validity and measure quality. However, a major change is made in the function weighting. It is felt that it is logically incorrect to assign equal weights to the functions and then attempt to rank the tests (each of which is primary with respect to one function). To eliminate this inconsistency, the functions are weighted logarithmically This arrangement is justified not only by professional opinion as to the nature of the ranking, but also because the biomedical and behavioral test weights are then equivalently distributed. The k value associated with each function measured by a given test is multiplied by the function weight to determine the W value. The confidence level associated with a set of tests is then found as before by summing the highest W values for each function listed under any of the measures included in the set.

The data relevant to the behavioral functions and tests is shown in matrix form in Table 39 and Table 40. The functions to be measured together with their weights are listed in the order of their weight magnitudes on the left side of the table The primary tests associated with each function are listed across the top of the tables. In many cases, the name of the test is the same as the name of the principal function that it is designed to measure. A block of the matrix has an entry whenever the test (identified by the matrix column) can measure the function (identified by the matrix row). The diagonal blocks obviously all have nonzero elements The number in each block of Table 39 is the k value of the test-function pair corresponding to that block. The number in each block of Table 40 is the weighted k value (i.e., the product of k and the weight of the function identified by that row). It is apparent that if all 52 tests are included in a set, the cumulative confidence value of the set is merely the sum of the weighted k values in all of the diagonal blocks (i.e., 0.8309).

The above data is now related to the measure of effectiveness of the behavioral experiments derived during the present study. The factors governing the selection of the behavioral experiments in the Apollo X missions are described in Volume 3 of this report. All 52 measures can be included in missions with a laboratory module. Thus, the confidence in the tests in

Table 39. Summary Data Sheet for Computation of Confidence Values for Behavioral Functions (Logarithmically Weighted Functions)

| Rank No | Name of Measured Function | Weight of Function | No. and Name of Measure | 1 Arm/Hand/Finger-Manipulation | 2 Arithmetical Reasoning | 3 Docking | 4 Reentry | 5 Complex Pattern Discrimination | 6 Mass Handling | 7 Mass Discrimination | 8 Don/Doff Suit | 9 Learned Procedure | 10 Vigilance | 11 Speaking | 12 Speech Perception | 13 Arm/Hand-Control of Force | 14 Detection of Angular Acceleration | 15 Detection of Linear Acceleration | 16 Reaction Time – Visual | 17 Arm/Hand/Finger-Control of Speed of Motion | 18 Detection/Discrimination of Force Against a Limb | 19 Detection of Limb Movement | 20 Recognition of Limb Location | 21 Compensatory Tracking | 22 Visual Resolution of Detail | 23 Depth Perception – Near | 24 Measure of Visual Fields | 25 Tone Detection | 26 Stereognosis | 27 Detection of Vibration | 28 Color Discrimination | 29 Fault-Phase Troubleshooting | 30 Data Recording and Conversion | 31 Time Estimation | 32 Depth Perception – Static, Far | 33 Depth Perception – Dynamic, Far | 34 Decision Making | 35 Cue Abstraction | 36 Problem Solving | 37 Scheduled Maintenance | 38 Detection of Light Touch | 39 Brightness Discrimination | 40 Association, Memory | 41 Reading Comprehension | 42 Graphoanetics | 43 Tone Pattern Discrimination | 44 Assessment of Volume in Space | 45 Sound Localization | 46 Pain Detection | 47 Detection of Heat/Cold | 48 Texture Discrimination | 49 Auditory Detection/Discrimination of Motion | 50 Detection of Tone Duration | 51 Olfaction | 52 Inductive Reasoning |
|---|---|---

Table 40. Summary Data Sheet for Computation of Condidence Values for Behavioral Functions (Logarithmically Weighted Functions

| Rank No | Name of Measured Function | Weight of Function | No. and Name of Measure | 1 Arm/Hand/Finger-Manipulation | 2 Arithmetical Reasoning | 3 Docking | 4 Reentry | 5 Complex Pattern Discrimination | 6 Mass Handling | 7 Mass Discrimination | 8 Dexterity/Deft Suit | 9 Learned Procedure | 10 Vigilance | 11 Speaking | 12 Speech Perception | 13 Arm/Hand-Control of Force | 14 Detection of Angular Acceleration | 15 Detection of Linear Acceleration | 16 Reaction Time – Visual | 17 Arm/Hand/Finger-Control of Speed of Motion | 18 Detection/Discrimination of Force Against a Limb | 19 Detection of Limb Movement | 20 Recognition of Limb Location | 21 Compensatory Tracking | 22 Visual Resolution of Detail | 23 Depth Perception – Near | 24 Measure of Visual Fields | 25 Tone Detection | 26 Stereognosis | 27 Detection of Vibration | 28 Color Discrimination | 29 Fault-Phase Troubleshooting | 30 Data Recording and Conversion | 31 Time Estimation | 32 Depth Perception – Static, Far | 33 Depth Perception – Dynamic, Far | 34 Decision Making | 35 Cue Abstraction | 36 Problem Solving | 37 Scheduled Maintenance | 38 Detection of Light Touch | 39 Brightness Discrimination | 40 Association, Memory | 41 Reading Comprehension | 42 Graphometrics | 43 Tone Pattern Discrimination | 44 Assessment of Volume in Space | 45 Sound Localization | 46 Pain Detection | 47 Detection of Heat/Cold | 48 Texture Discrimination | 49 Auditory Detection/Discrimination of Motion | 50 Detection of Tone Duration | 51 Olfaction | 52 Inductive Reasoning |
|---|---|---

this case is 0.8309. In missions without a laboratory module, tests 6, 14, 15, 19 and 20 cannot be performed. The confidence in the set of remaining 47 measures is calculated by the previously described process to be 0.7986 Thus the relative effectiveness of the command and service module configuration compared to the laboratory configuration is

$$\frac{0.7986}{0.8309} = 0.961$$

It is interesting to note that the weighting distribution does not change the total confidence level if all the tests can be performed (The small difference between the 0 80 cumulative confidence of the entire set as calculated in the BAHFR study and the value 0.8309 as calculated here is caused by changes in the original function list) However, if only a few tests can be performed the situation is quite different. Thus the greater weights assigned to the early ranked functions in the logarithmic distribution cause the cumulative confidence level to increase much more rapidly than before This increase is graphically shown in Figures 40 and 41 for the equal weighting (BAHFR) and the logarithmic weighting, respectively. In the latter case, a 70-percent confidence level is attained by performing only the first 17 tests. This procedure leads to the conclusion that almost any reasonably complete set of tests can satisfactorily determine the effects of weightlessness on behavior.

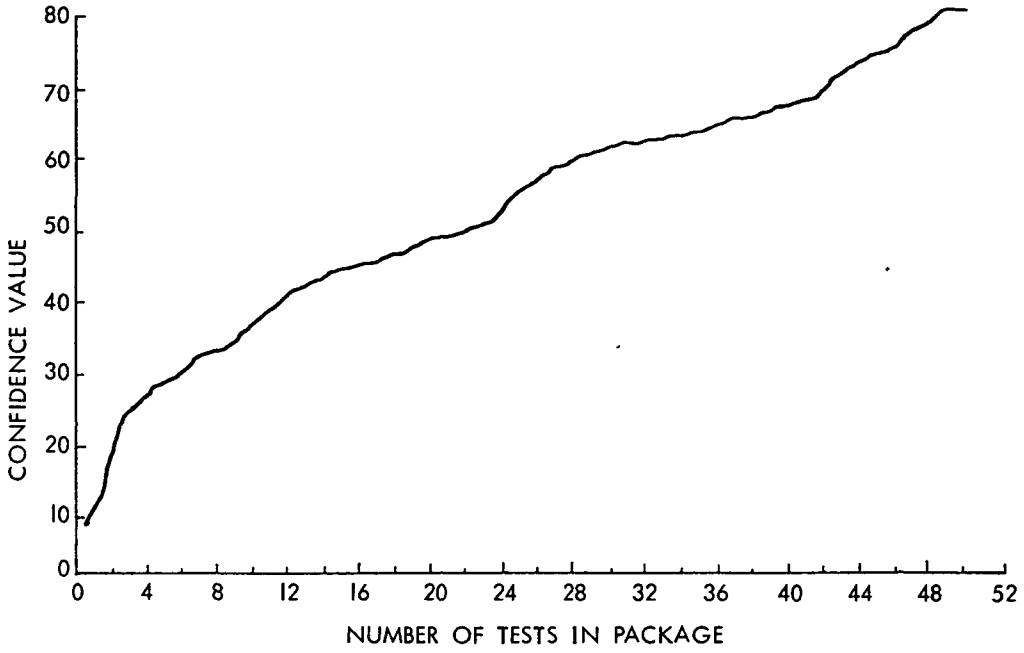

Figure 40. Cumulative Confidence Values for Behavioral Tests (Functions Equally Weighted)

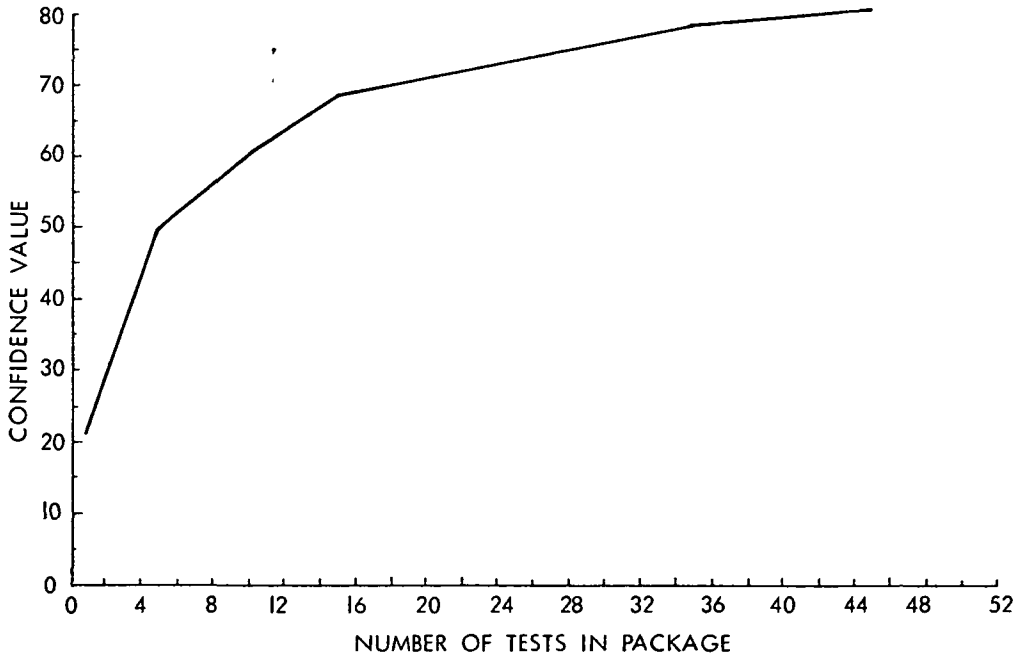

Figure 41. Cumulative Confidence Values for Behavioral Tests (Functions Logarithmically Weighted)

TYPICAL COST COMPARISONS

Two basic approaches were followed to arrive at Apollo X configuration designs for earth-orbiting missions as shown in Figure 42 . The first approach was to design an Apollo X, with and without a laboratory module, which would perform both earth-orbital missions and (with minimum modification) a lunar survey mission. Two alternatives existed (1) to let requirements for lunar mapping vehicle subsystems dictate earth-orbital mission duration (cases 1 and 2), and (2) to make minimum modifications in the lunar mapping vehicle subsystems to obtain capability for a 45-day earth-orbital mission (cases 3 and 4). The second approach was to make optimal (minimum) changes to the Apollo Block II command and service module configuration to enable it to perform the 45-day earth-orbital mission both with and without a laboratory module (cases 5 and 6). These six cases were selected for the cost comparison analysis.

The major objectives of the cost comparison analyses are (1) to determine the relative utility of the various vehicle configurations in the performance of certain priority experiments relating to the biomedical and behavioral effects of the space environment on men, (2) to determine the potential versatility for conducting typical scientific and engineering experiments, and (3) to determine the relative costs of the two design approaches (standardized vehicle for earth orbit and lunar survey missions versus an optimal change configuration from Apollo Block II for earth-orbital missions only) for both two-man command and service modules and three-man command and service module with laboratory.

BASIS FOR COMPARISONS

The cost comparisons were based upon the cost factors of available weight, volume, crew time, dollars, and experimental value. Results of the mission analysis and subsystems analysis were reviewed, and the major changes to the Apollo command and service module Block II configuration for the six selected cases were summarized to provide the basis for the cost comparisons. The major changes were defined in an absolute sense in terms of weight, volume, and complexity in order to determine dollar costs. (See Volume 22 of this report.) The resultant configurations for the six cases were then evaluated to determine the relative cost effectiveness of the various designs.

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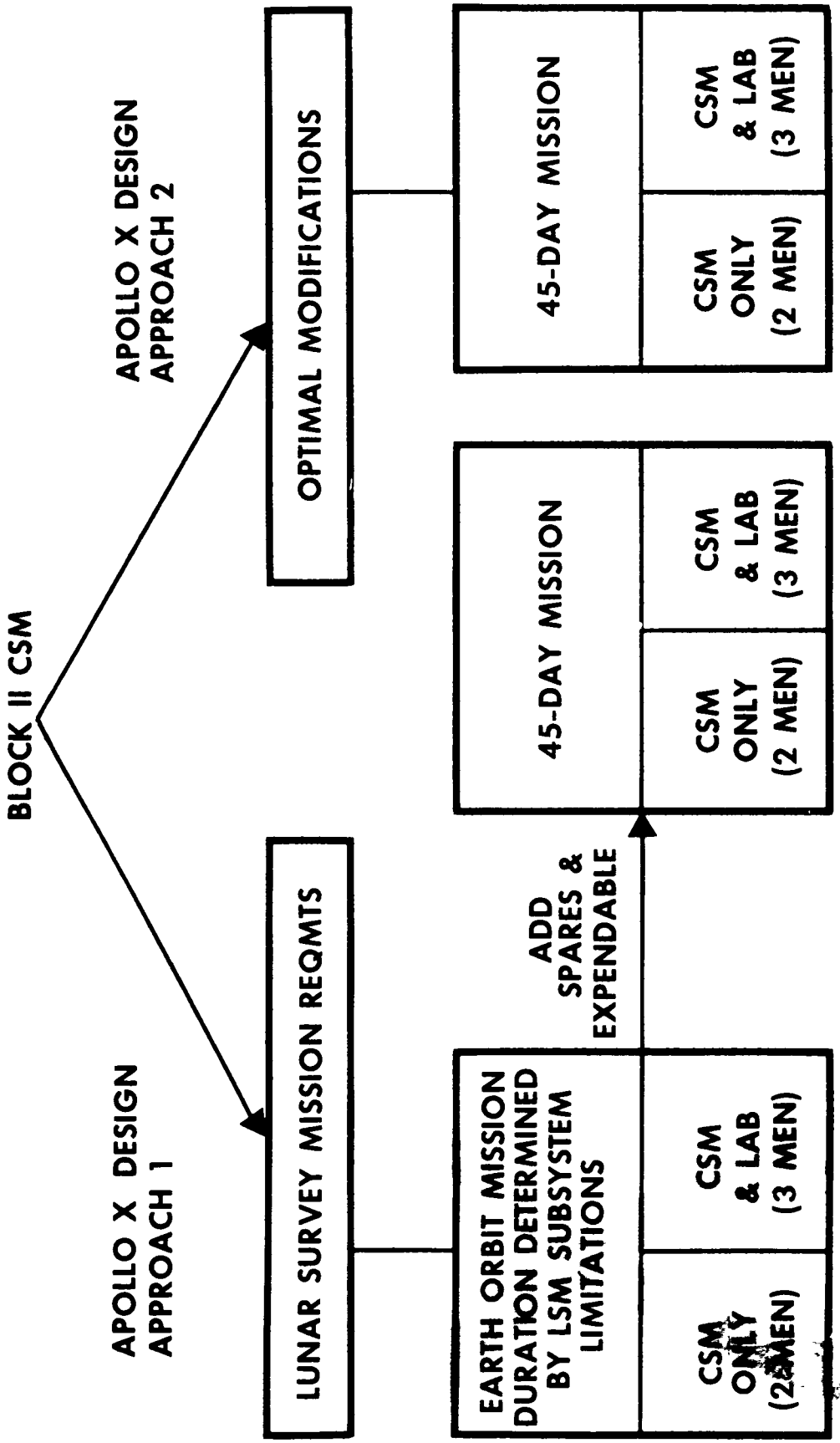
BLOCK II CSM
APOLLO X DESIGN APPROACH 1
APOLLO X DESIGN APPROACH 2
LUNAR SURVEY MISSION REQMTS
OPTIMAL MODIFICATIONS
EARTH ORBIT MISSION DURATION DETERMINED BY LSM SUBSYSTEM LIMITATIONS
ADD SPARES & EXPENDABLE
CSM ONLY (2 MEN)
CSM & LAB (3 MEN)
45-DAY MISSION
CSM ONLY (2 MEN)
CSM & LAB (3 MEN)
45-DAY MISSION
CSM ONLY (2 MEN)
CSM & LAB (3 MEN)

Figure 42. Cases Selected for Evaluation, 28.5-Degree Earth Orbit, BAHF Mission

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It is almost impossible in the absence of a well-defined space program (time-scheduled goals) to assign quantitative values to scientific and engineering experimental programs, because the value of a given experiment is to a large degree dependent upon the interests of the sponsoring agency. Because quantitative values of the confidence levels for the biomedical and behavioral experiments (BAHF) were available, however, the mission value analysis was limited to product-value comparisons for these BAHF missions in low-inclination earth orbits.

During the course of the mission and subsystems analyses, it was determined that the system lifetimes and reliabilities of the standardized lunar mapping vehicle were adequate for a 45-day low-inclination earth-orbital BAHF mission. Hence, the vehicle configuration for cases 1 and 3 and cases 2 and 4, respectively, are identical and thus the cost comparison analysis is reduced to four vehicle configurations.

It is evident that development planning costs are dependent upon the program approach (i.e., program costs for building an optimal change vehicle and then modifying it for the lunar mapping mission would be different from the costs incurred by building an initial standardized vehicle capable of both earth orbit and lunar mapping missions). Therefore, the dollar costs used in the cost comparison are the direct costs involved in modifying the Apollo Block II configuration for BAHF mission capability, within the constraints of the two design approaches.

System Changes Costed

The cost comparisons were based upon the following major system changes to the basic Apollo command and service module Block II configuration.

Cases 1 and 3

The design philosophy followed was to consider the technical requirements for the lunar mapping mission in addition to the earth-orbital mission requirements. This consideration resulted in a standardized vehicle configuration that, with minimum design changes, would have capability for lunar mapping. Major changes to the Apollo command and service module Block II consist of the following:

Command module

Install experiments

Redesign couch for two men

Rearrange main display panel

Provide food storage for 90 man-days

Remove G&N (retain optics package)*

Modify stabilization and control system

Remove one back-pack

Install two-gas system

Provide storage for additional LiOH canisters (60 required)

Provide hygienic supply storage for 90 man-days

Modify communications system

Provide crew systems storage (less food and back-pack)

*G&N system integration with the vehicle at a later time is provided for (i.e., provision in the design for harness installation, cooling and power requirements, and black-box mountings).

Service module

Remove rendezvous radar

Replace Block II SPS tanks with small spherical tanks

Add two fuel cells

Redesign cryogenic system (O_2, H_2, N_2) (metabolic and fuel cell requirements)

Replace Block II RCS tankage with two sets of LEM RCS tankage*

Add thermal control system for RCS system*

*These modifications are not required for the BAHF mission but are nevertheless incorporated into the standardized design for cost comparison purposes.

Case 5

The design philosophy followed was to consider the optimal changes to the Apollo command and service module Block II configuration for the performance of a two-man, 45-day, 28.5-degree earth-orbital BAHFR mission. The major differences between this case and the standardized design cases 1 and 3 are as follows.

Command module

No consideration is given to later installation of the G&N system in this case.

Experiment installation variations

Different locations for food and hygienic supply storage

Service module

The Block II service module RCS system is used in this case instead of the LEM RCS tankage. No RCS thermal control system is required.

Cases 2 and 4

The philosophy of standardized design is applied as in cases 1 and 3. However, a laboratory is added, and a three-man crew is employed. The major differences in design between cases 2 and 4 and cases 1 and 3 are as follows:

Command module

Three-man Block II couch assembly

Rearranged main display panel (minor modifications to Block II panel)

No food storage in CM

No experiment installations in CM

Storage for total of 4 LiOH canisters

Block II storage for hygienics supplies

Block II crew systems storage

Service module

Add one additional N_2 tank

Adapter

Remove adapter brace

Laboratory

Add small laboratory module

Install experimental equipment and supplies

Install LSS equipment and stores

Store LiOH canisters (86 required)

Extend ECS system

Install intercom to lab

Case 6

The vehicle configuration philosophy followed is identical to that for case 5, except that optimal changes to the Apollo command and service module Block II configuration are made to achieve a three-man, 45-day 28.5-degree earth-orbital BAFH mission utilizing a small laboratory within the confines of the LEM adapter. The major differences between this case and the standardized design cases 2 and 4 are as follows:

Command module

No consideration is given to later installation of the G&N system in this case.

Service module

The basic Block II Service Module RCS system is used in this case instead of the LEM RCS tankage. No RCS thermal control system is required.

TYPICAL PRODUCT VALUE VARIATIONS

As previously noted, the mission effectiveness measure for the cost analysis is based upon the BAHF experiments only, and variations in this effectiveness caused by other mission requirements were not determined. However, in addition to the product value of the BAHF experiments, data is presented which reflects the potential for performing other experiments. Table 41 contains data for the relative product value of the BAHF experiments and available experiment potential for the six selected cases.

The relative product value for the BAHF experiments is independent of the design approach used (standardized versus optimal design) but is weakly dependent upon the vehicle configuration(command and service module only or command and service module with laboratory). An increase of approximately 20 percent in product value is realized for the biomedical experiments when going from the command and service module to the command and service module with laboratory configurations. For the behavioral experiments, an increase in product value of approximately 3.7 percent is realized. The importance of these increases in the experiment

Table 41. Mission Effectiveness Comparisons

	CSM (Two-Man)			CSM and Lab (Three-Man)		
	34 Days	45 Days		34 Days	45 Days	
	Cases 1 and 3	Cases 1 and 3	Case 5	Cases 1 and 3	Cases 2 and 4	Case 6
A. Relative value of experiment product						
Experiment time (man-hours)						
Two-man earth-orbital BAHF	325	437	437			
Three-man earth-orbital BAHF				521	700	700
Number of subjects	2	2	2	3	3	3
Experiment (BAHF) confidence value						
Biomedical (maximum value = 1.0)	0.69	0.69	0.69	0.82	0.82	0.82
Behavioral (maximum value = 0.83)	0.80	0.80	0.80	0.83	0.83	0.83
B. Available experiment potential						
Total crew time (man-hours)	605	813	813	930	1247	1247
Volume-ft^3 (crew movement and experiment equipment)	142*	138*	138△	1341◇	1335◇	1335◇

*Volume for CM only; approximately 650 cubic feet available for experiments in the SM in sectors 2, 3, 5, and 6.

△Volume for CM only; approximately 711 cubic feet available for experiments in the SM in sectors 2, 3, 5, and 6.

◇Laboratory available volume; CM crew free area volume = 100 cubic feet; no experiments or food storage in CM; CM available storage volume of 16 cubic feet for experiments returned to earth.

product values is best evaluated by biomedical and human factors specialists and is not, therefore, discussed in this analysis. (See Volume 3 of this report for experimental programs.) However, they do not appear to be especially significant. The volume available for potential experiments includes total command module free volume plus volume available for service module-stowed experiments, which are accessible to the crew only via space. The differences in volume available for experiments in the service module for the various configurations are caused by the changes in RCS tankage sizes.

Although it is not shown in Table 41, an important factor that must be considered is the shape of the volume available for crew movement. In the two-man case, the removal of the center couch and stowage under the right-hand couch results in an additional free floor area of 8 square feet and a total unobstructed volume of working space of approximately 6 by 4. 5 by 4 feet.

COST COMPARISONS

Cost comparisons were made for the four basic vehicle configurations (cases 1 and 3, 5, 2 and 4, and 6) in terms of changes in weight, volume, and dollar cost from the Apollo Block II command and service module configuration. Only major changes costs were considered in order to show the relative magnitude of the costs of the various vehicles that can perform the BAHF missions. The major components considered for the cost effectiveness analysis are presented in Table 42. The differences from Apollo Block II costs include both recurring and nonrecurring costs. Hence, the total costs of development, manufacturing, testing, and program control are charged to one vehicle. Procurement of more than one vehicle would result in a lower amortized cost per vehicle.

Table 42 shows that the total differences from Apollo Block II costs are relatively independent of the design approach used. A difference in cost of approximately $3 million exists between the standardized design approach and the less expensive optimal design approach for both vehicle configurations (the command and service module only and command and service module with laboratory). The cost of the command and service module with laboratory configuration is significantly higher ($45 million) than that of only the command and service module configuration. This difference is due mainly to the cost of developing the laboratory, which has been assumed for study purposes to be $50 million.

Cost comparisons for the four vehicle configurations that can perform the BAHF mission are presented in Table 43 in terms of volume, weight, man-hours, and experiment product value. The following conclusions can be drawn from the results presented in Table 43.

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Table 42. Major Components of Cost Effectiveness Comparisons

Cost Factors / Subsystems	ΔApollo Block II											
	CSM Only						CSM + Laboratory					
	Weight (Pounds)		Volume (Cubic Feet)		$ - 10^6 Units		Weight (Pounds)		Volume (Cubic Feet)		$ - 10^6 Units	
	Case		Case		Case		Case		Case		Case	
	1 and 3	5	1 and 3	5	1 and 3	5	2 and 4	6	2 and 4	6	2 and 4	6
Add LEM adapter brace	+ 150	+ 150			0.005	0.005						
Add laboratory module									+1388	+1388	50	50
Experiments							+1068	+1068	+ 53.4	+ 53.4		
Crew LSS (less food and LiOH)							+ 115	+ 115	+ 25	+ 25		
Food storage							+ 256	+ 256	+ 9	+ 9		
LiOH storage							+ 386	+ 386	+ 13	+ 13		
ECS (extension to laboratory)							+ 101	+ 101	+ 2	+ 2		
COMM (extension to laboratory)							+ 35	+ 35				
Command module												
Remove one crewman	- 224	- 224	-10	-10								
Install experiments	+ 345	+ 345	+25.5	+25.5								
Couch redesign (Two-man)			-20	-20	3.0	3.0						
Redesign main display panel					1.1	1.1					0.200	0.200
Provide food storage	+ 72	+ 72	+ 1	+ 1	.763	.763	- 53.4	- 53.4	- 2	- 2		
Remove back pack	- 56	- 56	- 2	- 2								
Install Two-gas system (less LiOH)					3.0	3.0					3.0	3.0
Provide LiOH storage	+ 170	+ 170	+ 7.5	+ 7.5	.01	.01	- 79	- 79	- 4	- 4		
Remove G&N system (retain optics)	- 336	- 336	- 5	- 5	.088	.088	-336	- 336	- 5	- 5	0.088	0.088
Modify SCS	+ 44	+ 44			1.08	1.08	+ 44	+ 44			1.08	1.08
Service module												
Install experiments	+ 100	+ 100										
Replace SCS tanks with smaller tanks	-1197	-1197	-650	-650	.300	.300	-1197	-1197	- 650	- 650	0.300	0.300
Add two fuel cells	+ 757	+ 757			4.6	4.6	+ 757.5	+ 757.5			4.6	4.6
Redesign cryogenic system (O_2, H_2, N_2) (includes storables)	+2227	+2227			3.2	3.2	+3128.9	+3128.9			3.3	3.2
Replace Block II RCS tankage with LEM tanks	+ 439		+ 61		1.6		+ 439		+ 61		1.6	
Add thermal control for RCS	+ 141				1.5		+ 141				1.5	
Remove rendezvous radar	- 149	- 149	- 2	- 2			- 149	- 149	- 2	- 2		
	Δ Block II cost				20.246	17.146			Δ Block II cost		65.668	62.568

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Table 43. Cost Comparisons

Cost Factor	CSM (Two-Man) 45 Days		CSM + Lab (Three-Man) 45 Days	
	Standardized Configuration	Optimal Configuration	Standardized Configuration	Optimal Configuration
$ Cost (total difference from Block II)	20,246,000	17,146,000	65,668,000	62,568,000
Habitable experiment Volume, cubic feet	138	138	1,451	1,451
Total experiment Volume, cubic feet	788	849	2,101	2,162
Weight, pounds* (Limit for Experiments)	6,467	7,047	2,133	2,713

EFFECTIVENESS RATIO FOR VEHICLE CONFIGURATIONS

Cost Ratio	Volume	Experiment, Potential Weight	Experiment, Potential Weight	Experiment Product Value
Standardized Configuration ÷ Optimal Configuration				
Two-man	1 (man access) 0.93 (total)	0.92	1	1
Three-man	1 (man access) 0.97 (total)	0.79	1	1
Three-man/Two-man				
Standardized configuration	9.7 (man access) 2.7 (total)	0.33	1.4	1.11
Optimal configuration	9.7 (man access) 2.5 (total)	0.38	1.4	1.11

*Based on a maximum payload of 34,300 pounds

The effects of man due to prolonged confinement in a restricted volume and a weightless environment can be determined to a slightly higher degree of confidence when using the laboratory. A total experiment product value of 1.65 as compared to 1.49 for the command and service module only is obtained for the biomedical and behavioral experimental package out of a combined total possible value of 1.83.

Man-hours available for conducting experiments increase by a factor of 50 percent when the laboratory is used.

Crew habitable volume for conducting experiments when using the laboratory is ten times the volume when using the command and service module only.

The weight penalty incurred when using the laboratory limits the potential experiment payload to approximately one-third of the potential experiment payload for the command and service module configuration.

Mission cost effectiveness comparisons are presented in Table 44. The total spacecraft and launch vehicle costs were assumed or estimated follows

Configurations	Cost (Millions of Dollars)
Launch vehicle (assumed)	20
Basic (nonvariable) spacecraft costs	20
Delta (Variable) Spacecraft Costs (See Table 42)	
Cases 1 and 3	20
Case 5	17
Cases 2 and 4 (Assumed laboratory cost, $50 million)	66
Case 6 (Assumed laboratory cost, $50 million)	63

The resultant values for the cost effectiveness ratios were obtained by dividing the total spacecraft cost by the cost factor data of interest obtained from Table 41 and 43. Habitable experiment volumes of 138 and 1451 cubic feet for the CSM and CSM with laboratory configurations, respectively, were used. The costs per cubic foot of total available volume (including SM experiment volume) are based upon total volumes as shown in Table 43. Experiment payload liminations also vary with each vehicle

configuration as shown in Table 43. The costs per experiment-man-hour are based upon the total available crew time for conducting experiments as shown in Table 41. The costs per pound of the selected experiment packages are based upon the experiment package weights as shown in Table 42.

From the results presented in Table 44, it costs approximately 5 percent more to use a standardized design for the command and service module-only configuration and approximately 3 percent more (for the standardized design) for the laboratory configuration.

Table 44. Cost Effectiveness Comparisons

(45-Day Earth-Orbital BAIIF Mission)

	CSM Only (Two Men)		CSM With Laboratory (Three Men)	
	Standardized Configuration	Optimal Configuration	Standardized Configuration	Optimal Configuration
Total unit cost, ($' spacecraft + launch vehicle	60,000,000	57,000,000	106,000,000	103,000,000
Cost 'habitable exp vol ($/cubic feet)	434,800	413,000	73,000	71,000
Cost 'total exp. vol ($/cubic feet)	76,100	67,100	50,500	47,600
Cost subject day ($/day)	666,700	633,300	785,200	763,000
Cost exp. man-hour ($ man-hour)	73,800	70,110	85,004	82,600
Cost pound of experiments ($/pound)				
Selected package	134,800	128,100	99,300	96,500
Maximum available weight	9,300	8,100	49,700	38,000

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